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No. 716

DEVELOPMENT OF THE RULES GOVERNING THE STRENGTH OF AIRPLANES

By H. G. Küssner and Karl Thalau

PART I

GERMAN LOADING CONDITIONS UP TO 1926

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PART I. GERMAN LOADING CONDITIONS UP TO 1926

1. Introduction

The loading conditions together with the specifications regarding the classification of airplanes, the structural materials and their manufacture, the structural design of airplane body and power plant, the equipment and the service conditions, form part of the general strength specifications for airplanes. They serve as minimum requirements for the stress analysis of new types and are based upon the experience collected from older types during actual service.

The question of loading conditions for airplanes always has been a very contentious subject. For, aside from the airplane, there is no other vehicle of transportation in which the net weight is so decisive for its economy, in which it is so absolutely essential that the weight of its structural components be reduced to the lowest permissible minimum, and there also is no other mode of transport in which insufficient strength has such disastrous consequences.

The pioneers of aviation, such as Langley, Lilienthal, Wright, Ferber, Etrich, already had some ideas as to how the wings carried the load of the fuselage. They conceived this load as evenly distributed across the span and proportional to the strength of the wing in a manner such as to be able to support this load in safety. In those days the factor of safety (against failure) did not have to be high, because these first airplanes were flown very carefully and

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*"Die Entwicklung der Festigkeitsvorschriften für Flugzeuge von den Anfängen der Flugtechnik bis zur Gegenwart." Luftfahrtforschung, vol. X, no. 1, June 21, 1932, pp. 1-23.

for that reason they actually did not have to support loads much heavier than their own weight.

It is true that the Wright brothers attained a factor of safety of 5 against the normal weight, back in 1903, according to a letter sent to the French magazine L'Aerophile (reference 1).

Following Eiffel's and Prandtl's experiments with airfoils in the wind tunnel, in 1909, it became known that the air loads are not evenly distributed across the wing chord, and that a greater proportion of the load is borne by the leading edge of the wing.

That airplanes in turning or in gusts have to support greater loads than their own weight was learned from wing failures as soon as airplanes were able to fly higher and for longer distances, because here the probability of getting into unfavorable flight attitudes was naturally bound to be greater than in the first short hops.

Very illuminating from this point of view is the comparison of the record flights with the number of accidents (reference 2) appended in table I.

Table I. Records and Accidents - 1908 to 1913

	1908	1909	1910	1911	1912	1913
Speed (km/h)	65	77	109	133	174	204
Altitude (m)	100	475	3100	3900	5610	5850
Distance (km)	124	232	584	723	1010	1230
Number of airplanes	-	200	1300	-	-	-
Number killed	1	3	28	64	103	140
Number injured	-	43	70	32	54	127
Number of wing failures	0	0	7	6	10	10

(km X .62137 = miles) (m X 3.28083 = feet)

As a result, the question of stresses of airplanes in flight has ever since 1911, been a familiar topic of discussion in the technical literature. The "Commission de navigation aerienne" inaugurated sand-load tests for airplane wings in France in 1911. During the following years - 1912 to 1914 - Reissner, Hoff, Baumann, and others in Germany, undertook a systematic test program of airplane stresses, theoretical and experimental, which subsequently was taken over and continued by Hoff in the D.V.L. (German Experimental Laboratory for Aeronautics). The results of those investigations formed, in the war of 1915-1918, the basis of the load specifications for military airplanes, published by the Inspection Branch of the Flight Section.

Originally the multiple of the airplane weight supported by the wing in form of sand load up to rupture, was called the "safety factor," and subsequently, "load factor," and the strength specifications "loading conditions" as in structural engineering. Apprehensive because of the great number of accidents, the necessity of governmental supervision of airplane design became soon apparent. Beginning on October 8, 1910, a mixed commission, to which England, France, and Belgium subscribed, issued an "airworthiness certificate," which contained the regulations for static testing, flight testing, as well as general design specifications. As to the expediency of such instructions, which removed the responsibility of the builder to a great extent, considerable diversity of opinion prevailed at first. Instead of the schematically defined minimum load factor, it now required formulas for analyzing the outside loads to fit the particular characteristics of the pertinent airplane, and flight tests for the determination of the principal coefficients (reference 3).

Even in those days there were available for such flight tests, recording cable tensometers, extensometers, and accelerometers (reference 4).

But there were divers reasons why such experiments did not gain much favor as time went on and finally led only to several special figures of the load factor rather than to the anticipated analysis of the stress procedure. Apart from the defects of the test equipment, the main reason was the erroneous posing of the problems; one had expected to determine definite relations between the properties of the airplane and the maximum stress in flight.

Admittedly, there is a physical limit to the possible stress, which is characterized by the dynamic pressure in a dive, the maximum lift coefficient and the maximum control force. But this limit is ordinarily so high as to be out of consideration except for military and acrobatic airplanes. And the physiological limit of the stresses which a man sitting down experiences for a short period probably lies equally so high as to be scarcely worthy of notice. no! Lieutenant Doolittle experienced accelerations as high as 11 g without injury (reference 5). Bleriot conceded the limit of the physically bearable acceleration to be considerably lower and with that in mind favored a lower wing strength. (See section 3, p. 18.)

The aim, to obtain the required strength as exact function, is unattainable, first, because an economical minimum useful load is a prime requisite, and second, the physical and psychical qualities of the pilot, the atmospheric conditions, etc., cannot be predicted. It is only possible to establish correlations between the airplane characteristics, the magnitude of the stresses, and the frequency of their occurrence, and that on the basis of statistical data. And for such costly and troublesome experiments, the necessary leisure was lacking at a time when the constructive development was to the fore.

Thus it came about that up to now the strength of airplanes has been largely governed by specifications which, perforce, had to be limited to the schematic data of several minimum figures, especially of the load factors for the wings.

Notation

A	(kg),	load on front spar.
a	(m),	1. deflection of cable by cable tensometer.
	(m),	2. length of float.
α,		angle of attack relative to zero lift direction.
B	(kg),	load on rear spar.

b	(m),	1. tire width.
	(m),	2. boat "
	(m/s²),	3. brake deceleration.
β,		angle of the Vee.
C, C_o, C_1,		coefficients.
C	(kg),	frontal resistance in case C.
C_o	(kg),	frontal resistance of upper wing in case C.
C_u	(kg),	frontal resistance of lower wing in case C.
c_a,		lift coefficient; $c_{a\ max}$ maximum lift coefficient.
c_m,		moment coefficient.
c_{m_o},		moment coefficient of whole airplane for zero lift.
c_{m_o}'		moment coefficient for zero lift relative to 0.25 wing chord.
c_n,		coefficient of normal force.
c_{n_H},		coefficient of normal force of horizontal tail group,
c_{n_S},		coefficient of normal force of vertical tail group.
c_r,		coefficient of resultant air force.
c_w,		drag coefficient.
c_{w_R},		drag coefficient of fuselage.
c_{w_h},		drag coefficient at top speed in level flight.
c_{w_o},		drag coefficient of whole airplane for zero lift.
γ,		1. angle on cable tensometer.
	(kg/m³),	2. specific weight of air.

D	(m)	1. propeller diameter.
	(m)	2. tire diameter.
D_o	(kgm²),	torsional stiffness of overhanging surfaces.
d	(m),	diameter of control wheel.
E	(mkg),	energy absorption of landing gear.
e,		1. base of natural logarithm.
		2. load factor.
e_{sp},		load factor of tail skid.
ϵ		gliding angle; $\epsilon = c_w/c_a$.
η,		propeller efficiency.
F	(m²),	wing area.
F_H	(m²),	area of horizontal tail surfaces.
F_{HF}	(m²),	area of stabilizer.
F_{HR}	(m²),	area of elevator.
F_s	(m²),	area of vertical tail surfaces.
F_{sA}	(m²),	developed propeller disk area.
F_o	(m²),	overhanging area.
f	(m),	total elastic travel.
G	(kg),	gross weight.
G_F	(kg),	wing weight.
G_L	(kg),	landing gear weight.
G_R	(kg),	fuselage weight; $G_R = G - G_F$.
G_n	(kg),	static wheel pressure.
G_o	(kg),	weight empty.
g	(m/s²),	acceleration.
H,		1. expectancy.
	(m),	2. pitch of propeller; H/D = pitch.

h (m), height of drop in landing gear tests.

i (m), radius of inertia of buckling supports.

k, k_0, k_2, coefficients

l (m), 1. wing chord.
(m), 2. length of cable tensometers.
(m), 3. length under column load.
(m), 4. immersed length of float.

l_H (m), distance of c.p. of elevator from c.g. of airplane.

l_H' (m), distance of 0.25 wing chord to 0.30 horizontal tail surface chord.

l_S (m), distance of c.p. of rudder from c.g. of airplane.

l_Q (m), distance of c.p. of aileron from c.g. of airplane.

l_a (m), distance of front spar from leading edge.

l_b (m), distance of rear spar from leading edge.

M (mkg), moment.

M_H (mkg), moment about lateral axis of airplane.

M_S (mkg), moment about normal axis of airplane.

M_Q (mkg), moment about longitudinal axis of airplane.

M_o (mkg), moment of upper wing.

M_u (mkg), moment of lower wing.

m (kg s^2/m), airplane mass.

m_s (kg s^2/m), mass of a propeller blade.

m_R (kg s^2/m), airplane mass reduced to direction of shock.

N (hp.), horsepower.

n	(1/min),	propeller r.p.m.
n, n_A, n_B,		load factor.
n_R,		load factor of fuselage.
n_{aR},		load factor of fuselage, case A.
n_{aS},		"safe" load factor for case A; $n_{aS} = n_a:S$.
n_{cH},		load factor of horizontal tail surfaces, case C.
n_{cS},		load factor of vertical tail surfaces, case C.
ω	(1/s),	angular velocity of rotation of airplane on pull-out.
P	(kg), (kg),	1. loading. 2. breaking load of tire.
P_o	(kg),	maximum elastic force.
p	(kg/m²),	wing loading.
p_H	(kg/m²),	load per unit area of horizontal tail surfaces.
p_{HF}	(kg/m²),	load per unit area of stabilizer.
p_S	(kg/m²),	load per unit area of vertical tail surfaces.
p_R	(kg/m²),	air pressure of tire.
Q	(kg),	sand load.
$q = \frac{\varrho v^2}{2}$	(kg/m²),	dynamic pressure.
q_A, q_B, q_C,	(kg/m²),	dynamic pressure in cases A, B, and C.
q_a	(kg/m²),	dynamic pressure of gliding flight with extreme forward c.p.; $q_a \sim q_{min}$.
q_e	(kg/m²),	terminal dynamic pressure.

$q_h = \dfrac{\rho_o v_h^2}{2}$	(kg/m^2),	dynamic pressure at maximum level flight.
r	(m),	pull-out radius.
ρ	$(kg\ s^2/m^4)$,	specific weight of air.
ρ_o	$(kg\ s^2/m^4)$,	specific weight of air at sea level.
ρ_G	$(kg\ s^2/m^4)$,	specific weight of air at ceiling.
S,	(kg),	1. factor of safety. 2. cable force.
s	(m),	diameter of cable.
σ	(kg/m^2),	stress.
t	(m),	mean wing chord.
t_{HF}	(m),	elevator-stabilizer chord.
t_R	(m),	rudder chord.
t_o	(m),	mean chord of upper wing.
t_u	(m),	mean chord of lower wing.
U_o,		coefficient of gust stress.
v	(m/s),	flight speed.
v_a	(m/s),	speed, case A; v_{as} safe speed, case A.
v_e	(m/s),	terminal speed in dive.
v_h	(m/s),	maximum speed of unaccelerated horizontal flight near ground level.
v_l	(m/s),	landing speed.
v_r	(m/s),	cruising speed.
W,		probability or expectancy.
w	(m/s),	1. sinking speed at landing. 2. velocity of a vertical gust.
Z	(kg),	centrifugal force of propeller.
z	(m),	distance of elastic axis from leading edge of wing.

2. The Problem of Safety

Before proceeding to the strength specifications proper, it appears necessary to elaborate upon the question of safety.

Safety in aviation is generally interpreted as the minimum possible frequency of accidents per unit of flight distance or flight endurance. As seen from table II, the frequency of accidents per flown mile has dropped materially during twenty years of technical progress. As a matter of fact, the frequency in German air transport accidents has almost dropped to that of automobile traffic (reference 6).

Among the flight accidents, wing failures play a particular role, not only because of the catastrophic consequences for the particular airplane, but also because of the deterrent effect on the public at large.

Table II. Accident Statistics (reference 7)

	kilometers flown in millions	number of accidents	airplanes destroyed	accidents involving fatalities and injuries	wing failure in flight	Fatalities		Injuries	
						total	wing failure	total	by wing failure
All countries 1909-1910	0.49	148	-*	-*	43	31	17	117	26
Germany, commercial 1926-1931	54	607	46	59	4	59	16	112	9
Germany, sport and training 1926-1931	35**	2007	296	274	19	123	17	269	7
U.S. commercial 1928-1930		1790	455	307	22	303	-*	673	-*
U.S. sport and training 1928-1930	460	2865	1293	973	40	741	-*	1255	-*

*No data available.
**By assuming a mean speed of 120 km/h (74.5 mi./hr.)

In German air traffic, for instance, only 0.66 percent of all failures in flight were wing failures, but they produced 27.2 percent of the fatalities. During 1909 to 1910, 55 percent of the fatalities were caused by wing failures, that is, more than half.

There is, however, despite what may seem depressing at first sight, an optimum lower limit to this quota; because an excessively strong and therefore heavy airplane can either no longer fulfill its purpose - carry useful load - or else is endangered again by the higher speed at landing and take-off, as well as by the greater exertion of the engine in flight. Since the final aim always will be to bring the total accident expectancy to a minimum, a certain amount of wing failures which, however, can be less than heretofore, will have to be counted with as inevitable even in the future.

By safety in the narrower sense is meant, in the following, a certain very small probability W_o of wing failure. This probability is for itself again the sum of probability of material, manufacture, design, and construction errors W_m, as well as of excesses W_b, of the design strength by extraneous forces caused by faulty pilotage, gusts, or vibrations. When flight was in its infancy $W_m \sim W_b$ (reference 8), in contrast to $W_m < W_b$ at the present, thanks to conscientious material inspection and supervision of manufacture. Thus we put $W_o \sim W_b$ in first approximation and disregard any eventually existing choice in the figures of the extraneous physical influences. Then the expectancy H, with which extraneous forces reach in unit time the load factor n, follows Gauss' law of distribution (reference 9):

$$H = C\ e^{-k(n-1)^2} \tag{1}$$

Constants C and k are determinable when the expectancy of two load factors n_1 and n_2 is known. The curve a in figure 1 is the expectancy of the extraneous forces (load factor n) versus the average life of an airplane (1,000 flight hours) by an assumed probability of failure $W_b = 10^{-3}$ by load factor $n_1 = 4.5$ and expectancy $H = 10^3$ of attaining the load factor $n_2 = 2.5$. These figures are applicable to wing stress only and correspond to those reached in service. The course of curve a, which for the time being is amenable only to estimation, must later be more accurately defined from statistical data, which were begun in 1931 on a large scale in the D.V.L. on the basis of the special ground work of E. Seewald and the writers (reference 10).

It is impossible to give any reliable information about the course of the expectancy curve until continuous measurements over a longer period are available. One single flight can, owing to propitious weather conditions and expert piloting, become very misleading (see curve a_1, which was shifted parallel conformable to the duration of measurement). Curves b and c are the fatigue strength in bending of pine wood and light metal (reference 11). It may be assumed that a test bar subjected to the actual, fluctuating stresses in an airplane, has a strength greater than the fatigue strength.

According to the latest investigations, on the other hand, the fatigue strength of built-up parts, such as wing spars, is markedly lower than that of test bars, so that the curves b and c may be looked upon as a clue to the true course of the fatigue strength. Because of the vitiating effect of abrupt cross-sectional changes, the fatigue strength of metal spars, for instance, can drop after about 10^6 reversals to 15 to 30 percent of the fatigue strength of test bars (reference 12).

Below the expectancy $H = 1$, the strength may be considered as equal to the static breaking strength. The strength of structural components which are dimensioned for stability failure, curve d in figure 1, is practically unaffected by the figure H of the stresses. The safety diagram (fig. 1) reveals that by the present short life of airplanes (about 1,000 flight hours on an average, although some airplanes of the Luft Hansa have reached as high as 3,000 flight hours), a one-time occurrence of a high stress is decisive for the probability of failure (cut a,d). The load factor 3.6, or 80 percent of the load factor at failure corresponds to the relative expectancy $H = 1$. Should the yield limit of the material be below this figure and permanent deformation be held imminent, which would preclude any further use of the airplane, the parts originally dimensioned for stress failure would have to be so strengthened that the yield limit is not appreciably exceeded up to load factor 3.6. In accord with the new strength specifications the yield limit of the material must therefore not be reached at less than 75 percent of the ultimate load in Germany and France, and at 100 percent in Holland and England. With longer life or especially unfavorable fatigue strength, i.e., parallel shifting of curve a in direction of the abscissa, cuts of curves a,b or a,c are also possible, in which case the probability of failure then becomes dependent upon the fatigue strength.

Another certain probability of failure exists when curves a and b approach each other, for they do not represent exact functions but rather averages of point clusters with a certain range of scattering. As protection against this, one can either strengthen the parts designed for failure in stress or else limit the life span of the airplanes, so that for the majority of the structural parts the probability of failure W_b remains equal to the probability of the one-time occurrence of stresses which exceed the static strength at failure.

Inasmuch as in light constructions the parts designed for stability failure (sagging and buckling), predominate as a result of the split-up method of construction and the thin wall thicknesses, the question of static breaking strength in airplane statics is, moreover, of primary interest. The parts designed for failure in stress form, on a weight basis, a smaller quota and are, for the reasons stated above, and in order to make the failure of the whole structural assembly independent of the unreliable breaking strength, especially freely strengthened. A light construction of this kind possesses the valuable quality of being able to withstand several loads up to close to the limit of failure without injury.

Exceptions hereto are such structural parts as engine mounts which, in normal service are exposed to enormous vibratory stresses, the expectancy of which may attain to $H = 10^8$ during the life of the airplane. In the face of such high figures, although common enough in machine design, the fatigue strength of test bars and still more that of built-up members, assumes very low figures, which then alone decide the probability of failure.

It was owing to the lack of methods for obtaining more appropriate material characteristics, that the breaking strength gained its foothold as reference quantity. The quotient of breaking strength and permissible theoretical stress was called "factor of safety." But the frequently astonishing height of this factor should not let one forget that the difference between the calculated safe stress and the true fatigue strength of the structural part can still be low in spite of it under service conditions, and that the height of the safety factor is rather an admission of the utter unfitness of the breaking strength as reference quantity, for the ignorance of the true stresses, and for the inaccuracy of the stress analysis. And so the breaking strength has at last been superseded by the yield limit, i.e., the stress at which 0.2 percent permanent elongation of the test length is reached.

For all component parts stressed predominantly by static as well as dynamic loads, which are designed for stability failure, the yield limit has proved the appropriate reference quantity. In some cases, as for wagon springs, for instance, the introduction of a safety factor became altogether superfluous. In bridge design the safety factor lies between 1.7 and 1.8 against exceeding the yield limit and against collapsing of short column members. When one considers that on railway bridges only very small permanent deformations for less than 0.2 percent are permissible, that the load variations, even if minor, can fatigue the material during the long life span of a bridge and that finally sectional and structural changes due to corrosion are inevitable, there remains a psychological, i.e., mathematically unfounded safety factor, a figure which, if at all, differs only slightly from 1. So, when the calculation is based on high stresses of low expectancy, in bridge design, for example, wind velocities of from 45 to 60 m/s (148 to 197 ft./sec.), and when all service conditions are allowed for, the introduction of safety factors could be omitted. Hence the insistent pressure which is being brought to bear on the elimination of a safety factor, which is merely a stop-gap in favor of the more rational dimensioning for probability of failure (reference 13) somewhat as shown in figure 1.

When we reflect on the results of modern statistical data and probability analyses in many branches of science, hitherto inaccessible for causal correlations, it is justifiable to anticipate marked progress in strength analysis also, and so increase the actual safety, rendering the factor of safety, which more rightly belongs in the ambit of psychology, unnecessary (reference 14). Moreover, the ultimate strength in airplane statics being itself of primary significance, considerations of probability of failure are here particularly in place.

The particular position of airplane statics has not always been clearly recognized. Following the number of wing failures in 1910-1911, there was no lack of recommendations for a safety factor of from 2-3 against the maximum occurring stresses and for ultimate load factors up to 12 (reference 15). These proposals could not be realized because with the then existing airplane types they were equivalent to a donation of a prize for useful load. Up to 1926 the general rule was to figure with the ultimate load factor without specially distinguished safety factors. And during the great war the ultimate load test of a part proved a reliable and economical check.

As the number of airplane types increased, the number of pieces of each series decreased and the now uneconomical ultimate load test was superseded by the static stress analysis along the lines of bridge design statics (reference 16). The tendency mentioned above, of strengthening the structural parts originally designed for failure in stress being of more recent date, one of the first obstacles encountered in the analysis was that the tension members in the neighborhood of the ultimate strength no longer followed Hooke's law, upon which the conventional bridge design statics were based. Since for most materials Hooke's law is still applicable by a stress equal to half the breaking strength, the practice was to carry the static analysis through to 50 percent of the breaking load, but on the other hand, to demand a safety factor of 2 against failure as the strength of each individual structural component.

Other than that the agreement between stress analysis and ultimate load experiment is often surprisingly close, the sources of error are, in principle, the same when

a) the proof of the stress is carried through for a service loading in which the assumptions of classical statics are still rigorously applicable and accordingly the breaking strength of each separate constituent is estimated, although properly this should only be effected in connection with the whole structure;

b) under the same premises which, of course, then, are invalid for part of the structural member, it is proved that all parts sustain the minimum breaking load demanded of the airplane.

The first method is preferred in bridge design because no stresses are likely to occur which come anywhere near the breaking load. A bridge is even less than an airplane to be considered a structure of equal breaking strength. Contrariwise, stresses approaching or exceeding ultimate load, are rare for airplanes in free flight, although much more probable than in bridges. Since the decisive empirical figure is the breaking load factor, while the safety factor 2 was set up conformably to the approximate ratio of $\sigma_B : \sigma_{proper}$, method b) is generally preferred in airplane statics, and has recently been improved by the so-called stress-strain laws (reference 17). Admittedly, this analysis of failure must be effected by fulfillment of a stipulated yield limit.

The loading conditions of several countries still contain the division of the ultimate load factor into an apparent or supposedly "safe" load factor and a safety factor of 1.8 to 2, conformably to method a). The previously known concept, "safe" load factor (reference 18), has the advantage of affording the designer legal protection, inasmuch as it only requires him to prove an exact stress up to stresses below a certain limit while permitting him to estimate the stress beyond this limit in accordance with the conventional methods of calculation.

It was emphasized that the calculation then conformed to conventional practice and it was also assumed that 2 as safety factor would be ample against material defects, inaccuracies in manufacture, as well as against short, abrupt load excesses. Logically "short, abrupt load excesses" should be excluded from the safety factor of airplane statics, if it really were to correspond to the safety factor in bridge design statics. But accident investigations have shown that even minute defects in material or its manufacture may lead to accidents and the very fact that the majority of wing failures revealed no such defects despite the most searching investigation, leads one to believe that the stresses must actually have exceeded the theoretical breaking strength.

Any appreciable probability of material or manufacturing defects in the main supporting members is, for air traffic, at any rate, untenable. And these defects become consistently more rare as the D.V.L., for example, has proved by its ultimate load tests since 1913. Even a reduction in strength to 3/4 of the theoretical strength would, according to the safety diagram (fig. 1) very probably lead to failure, because the anticipated expectancy of this stress is about 4 during the life of the airplane (reference 19). It remains therefore for the airplane manufacturer to take over the obligation for the theoretical minimum breaking strength by appropriate shop inspection and to take precautionary measures for its maintenance even in protracted service.

One remarkable feature is that the very countries in which this assumption is not at all, or only partially complied with, specify higher ultimate load factors for the same types of airplanes.

It was also attempted to give the auxiliary term of "safe" load factor some physical meaning in the sense of a superior limit. As is known, metal test bars undergo, when stressed beyond the elastic limit, elongations in increasing measure, which remain after unloading. The elastic limit (or proof stress) lies at practically half the breaking strength. Because of the fact that no permanent deformations - or if so, only to a slight degree - have been observed on metal airplanes in service, it was believed to be justified in concluding that the 1/2 breaking load factor is not at all, or only very rarely exceeded. But this presumption is contrary to the actual behavior of airplane wings, which is more propitious than that of a test bar for the reason that an airplane wing is, for manufacturing reasons, no perfect body of equal strength. The destructive load tests on wood and metal wings carried out by the D.V.L., revealed proportionality of outside forces and total deformations up to 80 to 90 percent of the breaking load (reference 20) and frequently, a surprisingly close agreement between the theoretical and the experimental breaking load. According to figure 1, the expectancy of such high stresses may remain < 1 during the whole life of an airplane, with the result that in most airplanes no permanent deformations are observed even when the mathematical expectancy of reaching the "safe" load factor 2.5 of the present diagrammatical example, amounts to 1,000. Experience with failures within the last few years together with protracted wing-deflection measurements, however, leave no doubt that the "safe" load factor represents no superior limit of stresses encountered in service. The safe load factor may be rather looked upon as a stress still well reproducible in flight test, whose average expectancy during one hour of flight in the above example was accepted at $H = 1$.

The introduction of "safe" load factors and of safety factor 2 was contemporary with a marked upswing in air traffic in Germany, France, Holland, and Italy, so that, aside from the legal aspects already cited, psychological reasons may possibly also have acted in favor of the "safety factor." But to-day the number of purely technical exigencies for a division of ultimate load factor into "safe" load factor and "safety factor" has dropped considerably. It is to be expected that the aims to reduce the probability of failure, i.e., raise the actual safety, begin with the exploration of the expectancy of the stresses and their extrapolation by Gauss' distribution law up to failure, and thus become free, at least physically, from the concept of safety factor.

In many cases it was found expedient to make certain parts of an airplane stronger or weaker than others adjacent to it; that is, to grade its probability of failure. Thus, it is a rule to figure the fuselage with a higher safety factor than the landing-gear struts, the latter with a higher safety factor than the wheel axle, or generally, tension members for greater safety than members stressed in compression. The analogy that "the chain is no stronger than its weakest link" is not always applicable to the airplane because the stresses impressed upon it on the ground are vastly different from those encountered in the air.

Since it is important to make the total probability of accident a minimum and at the same time insure a minimum structural weight, special reenforcements are applied wherever any marked reduction in breaking or accident probability can be effected by a small increase in weight. For illustration, it would serve no useful purpose to build an airplane wing stronger when the considerable additional weight necessary to accomplish it, if applied to other vital parts, such as tail or controls, would bring about a much greater reduction in total accident probability. The accident probability is especially great in landing gears and float supports, even though the results of such accidents are seldom fatal. There one attempts to control the sequence of failures by attenuating individual parts, and in that manner, protect the passengers as much as possible against injury. All these problems still await scientific treatment by means of statistical research.

3. Start of Development

France was the first country to realize the importance of the airplane, and the French Army acquired a number in 1910. The experience gained from testing these airplanes resulted in the establishment of a minimum breaking-load factor n for army airplanes, then called "indice d'essai statique" (static test index) (reference 21).

Year	1912	1913	1914
n =	3	3.5	4.5

The stresses of an airplane in vertical gusts and by pull-out from a dive were first analyzed by P. James (reference 22). Bleriot proved that wing failure due to pressure from above is possible when sharply changing from level into gliding flight; however, he cautioned against any exorbitant load factor because the accelerations to which a man, sitting down, is subjected, do not exceed 5 to 6 g (reference 23). W. Voigt computed the possible stress by pull-out from a dive (reference 24) at

$$P = 0.1\ v^2\ F \tag{2}$$

Delaunay compared the stress when flying into a horizontal gust (reference 25), with that in undisturbed flight and obtained as stress ratio $(v + \Delta v):v$. From this he deduced that the fastest airplane utilizes the structural weight most evenly. Clarke investigated the maximum wing stress by pull-out from a dive (reference 26). He then integrated the flight path equations and arrived at the result that after a 300 m (985 ft.) dive the load factor 9.5, after a longer dive a load factor 11.5, is obtainable when the elevator is suddenly displaced.

These and other related problems were exhaustively treated at the safety meeting of the Permanent Commission for International Aeronautics, during its October 4-6, 1912 sessions (reference 27).

Public interest in aviation in Germany found expression in the formation of the W.G.L. (Scientific Society for Aviation) and of the D.V.L. (German Experimental Laboratory for Aeronautics) in 1912. Although the purely aerodynamic problems had engaged Dr. Prandtl's attention since 1908, in the small model experimental station at Göttingen, public interest now was ripe for the many other problems in flight technique. The stress and safety of airplanes formed the subject of H. Reissner's report at the First General Meeting of the W.G.L., on October 25, 1912 (reference 28). Its importance warrants an analysis.

Table III. German Reliability Contest, 1911

Airplane type		Aviatik biplane (Farman)	Aviatik biplane (Farman)	Albatros biplane	Euler biplane	Rumpler monoplane	Wright biplane	Dorner monoplane
G_0	kg	450	480	350	280	500	350	400
G_{full}	kg	629	658	543	455	669	449	575
F	m^2	45	60	50	30	32.5	35	26
p	kg/m^2	14.0	11.0	10.9	15.15	20.5	12.8	22.1
N	hp.	100	70	70	50	60	52	41
G/N	kg/hp.	6.29	9.4	7.8	9.1	11.15	8.6	14
v	m/s	32	22	22	22.3	25.5	23.5	22.3
ϵ	$= \frac{45\ N}{vG}$	0.224	0.217	0.263	0.222	0.158	0.221	0.144
c_a	$= \frac{17\ p}{v^2}$	0.233	0.387	0.383	0.520	0.537	0.394	0.756
c_w		0.052	0.084	0.101	0.115	0.085	0.087	0.109

(kg × 2.20462 = lb.) (kg/m² × .204818 = lb./sq.ft.)

(kg/hp. × 2.17442 = lb./hp.) (m/s × 3.28083 = ft./sec.)

To visualize the notions of those days, we reproduce a 3-view drawing of the then fashionable Albatros biplane of 1911 (reference 29). The wing had two spars, a front spar on the nose and a rear spar at 3/4 wing chord; its camber was 1:15 (fig. 2). The performances obtained with one of this type along with six other entries, in the 1911 German Reliability Contest, are tabulated in table III.

The lift (c_a) and drag (c_w) coefficients are computed for an assumed propeller efficiency of 0.60, and for an air density of $\rho = 0.118$ kg s^2/m^4 and plotted as + in the polar curve of figure 3. One noteworthy feature is that the airplanes at that time were not much inferior to the modern commercial airplanes despite what, according to modern conception, appears as quixotic shapes, as a comparison of the + points in figure 3 with the polar curve of a much later commercial airplane (fig. 4), reveals. Reissner based his calculations on the Prandtl-Föppl polar curve for a rectangular thin curved plate (curve a, fig. 3), and added $c_{w_R} = 0.138$ for parasite drag of fuselage and control surfaces. In this manner he obtained polar curve b for the whole airplane. Its drag coefficients are about twice as high compared with the test points + for full scale. This was probably due to the then little-known effect of the Reynolds Number of model testing.

Reissner's report first treats of the effect of the structural airplane weight as most important of the outside loads. On cambered airfoils the moment about the leading edge of the wing (coefficient c_m) does not change proportionally to the lift; as a result the c.p. shifts when the angle of attack is changed and thus stresses the wing in varying fashion. For a two-spar wing of the above-described type (fig. 2) the equations of the normal spar loads (fig. 5) are:

$$\left.\begin{aligned} A + B &= \frac{\rho v^2}{2} F \left[c_a \cos\alpha + c_w \sin\alpha\right] \\ A\, l_a + B\, l_b &= \frac{\rho v^2}{2} F\, l\, c_m \end{aligned}\right\} \qquad (3)$$

In steady gliding flight under force of gravity, the net airplane weight is

$$G = \frac{\rho v^2}{2} F c_r ; \quad c_r = \sqrt{c_a^2 + (c_w + c_{w_R})^2} \qquad (4)$$

The normal forces with $l_a = 0$, $l_b = 0.75\, l$ are in this case:

$$\left.\begin{aligned} \frac{B}{G} &= \frac{c_m}{0.75\, c_r} \\ \frac{A}{G} &= \frac{c_a \cos\alpha + c_w \sin\alpha}{c_r} - \frac{B}{G} \end{aligned}\right\} \qquad (5)$$

The maximum possible stresses in the wing spars in still air are obtained, after a suggestion by Von Parseval, when the airplane is pulled out at the highest possible speed without loss of speed by rapid elevator displacement. The highest speed is obtained as permanent state of a continuous dive and follows from equation (4) after inserting the minimum value for c_r.

For the case of steady gliding flight and pull-out from a dive the normal forces on the spars, given in table IV, are obtained.

Table IV. Normal Forces Acting on Wing Spars

α		-3.8°	-1.8°	0°	2.5°	5°	10°	15°	20°
Polar curve b	c_a	-.092	.088	.300	.516	.820	1.146	1.266	1.05
(fig. 3)	c_r	.226	.216	.358	.554	.850	1.180	1.328	1.166
	c_m	.050	.111	.205	.270	.372	.432	.469	.455
Glide	A:G	-.713	-.281	.080	.285	.389	.488	.500	.417
(Reissner)	B:G	.283	.684	.761	.649	.584	.488	.471	.502
Dive									
(v. Parseval)	A:G	-.83	-.31	.15	.81	1.71	2.98	3.43	2.56
($c_{r\,min}$=0.194)	B:G	.34	.76	1.39	1.86	2.56	2.98	3.23	3.20

In a steady glide the highest stresses in the spars occur at low lift coefficients. At $c_a = 0.3$ the rear spar (B) has to take up 0.76 of the total load. The speaker therefore recommends the analysis of this - later called "loading condition B," - attitude of flight. The c.p. is approximately at 2/3 of the wing chord, the speed is about twice the landing speed, not of the normal. The inclusion of a 33° gliding angle in this attitude of flight is necessary because of a too high estimate of the drag coefficient. Apart from that, the airplanes at that time were able to reach a lift coefficient $c_a = 0.3$ even in level flight by full throttle. (See table III.) This flight attitude was subsequently adopted to represent case B, especially in foreign countries.

In pull-out from a dive the stress in the two spars is highest when $c_a = c_{a\,max} = 1.27$. The c.g. is then about 1/3 of the wing chord. This later became "load case A." The speaker, however, believed this to be too unfavorable because of the presumed constant speed while the airplane changes from 0 to 15° angle of attack. The overloads should be accorded much more weight at low lift coefficients and the rear spar be considered as the point of danger.

The production of the necessary elevator displacement for rapid recover without loss of speed is merely a matter of applied manual effort and so much more readily obtained as the airplane is smaller. Although stunt flying at high speed was then a rarity, it became quite frequent a few years later. The normal forces calculated for diving therefore represent the superior stress limit for those airplanes. (Compare the dashed line (- - -) of the highest possible stress in fig. 42.)

Instead of this most unfavorable case the stress of the airplane is analyzed as it flattens out from a steep glide. The radius of flight path is to be $r = 136$ m (446 ft.) by an assumed leveling-off height of 40 m (131 ft.) and a path angle change of 45°, the flight speed $v = 40$ m/s (131 ft./sec.). The acceleration in terms of multiples of accelerations g acting on the airplane at point A of the flight path, is

$$n = 1 + \frac{v^2}{rg} = 2.20 \tag{6}$$

Later it became customary also to express the ratio of normal acceleration acting on the airplane to acceleration due to gravity as "load factor n." In particular, it was assumed that the highest possible wing stress was n times the stress in steady flight, an assumption which does not always hold good by rapid changes in normal force.

During the transition from level to steep gliding flight the wing can be impressed by pressure from above, in which case the load factor $n = -1.0$ is at point D of the flight path. (See fig. 6.)

When an airplane describes a steady circular path in a horizontal plane whereby the wing axis slopes at angle β to the horizon, the load factor is $n = 1 : \cos\beta$. The limiting case in practice is $\beta = 45^\circ$ and the load factor is $n = 1.41$. In gusty weather the airplane is subjected to stresses due to changes in speed and in wind direction, regarding which no reliable data were available in 1912. Assuming constant airplane speed v the load factor in a horizontal gust with $\Delta v = 5$ m/s (16.4 ft./sec.) fluctuating velocity is

$$\left.\begin{aligned} n &= \left(1 + \frac{\Delta v}{v}\right)^2 \\ v = 20 \text{ m/s} &: n = 1.56 \\ v = 40 \text{ "} &: n = 1.27 \end{aligned}\right\} \tag{7}$$

According to that, it looks very much as if faster airplanes suffer lower stresses in gusts.

The corresponding formula for directional changes of the wind (vertical gusts) would yield a converse result, that is, higher stresses for faster airplanes. The speaker merely intimates that a load factor $n = 2$ appears obtainable in a directional change of the air current equal to the steady wing incidence, and recommends systematic and reliable measurements to decide all these problems. But in the meantime, that is, until such data are available, he proposes the load factors:

$n = 4$ to 6 for upward acting forces and

$n = 1$ for downward acting air loads.

For the stress analysis of wing spars a uniform loading toward the outside is recommended as a safe assumption. The loading across the wing ribs is very unevenly distributed, according to Eiffel's measurements. The safest way is to figure with a triangular load area whose apex is midway between the two spars. (See fig. 5.)

The maximum pressure on the control surfaces is

$$p = \frac{\rho v^2}{2} c_{n\ max} \sim 0.075\ v^2\ \frac{kg}{m^2} \tag{8}$$

which is reached at approximately $\alpha = 30^\circ$. But this formula is deemed too unfavorable, because the airplane turns immediately under the pressure of the control surfaces, so that the actual angle really does not exceed 15°. Then the control surface loads are of the order of wing loads.

Up to now the discussion has been confined to the components of the air loads normal to the wing chord. The loads in direction of the wing chord are very small. Nevertheless, practice has proved a substantial cross bracing of the wings as expedient, especially when contacting with the ground.

The engine mount is not stressed so very much by the propeller thrust, which amounts to 3.0 to 3.6 N^{hp} (kg), as by the engine vibrations.

Since poor terrain and bumpy air quite often result in rough landing, one should not rely upon the pilot's skill but rather figure with an angle $\epsilon \sim 0.1$ to 0.2 of flight path to ground level.

The kinetic energy of the landing shock shall be absorbed by the work of the shock absorber

$$E = \frac{mw^2}{2} = k\ f\ P_o \tag{9}$$

The factor k is contingent upon the elastic characteristic. When the elasticity rises proportionally to the travel (elastic shock absorption without initial tension), then $k = 0.5$; when with initial tension, k increases and becomes in the extreme case $k = 1$, which, in fact, even the old hydraulic shock absorbers reached fairly closely. Putting $k = 1$, an assumption ordinarily not eventuating by the then existing shock-absorber types, even with considerable initial tension, the impact factor is

$$e = \frac{P_o}{mg} = \frac{w^2}{2fg} \tag{10}$$

Conformably for a sinking speed $w = 4$ m/s (13.1 ft./sec.) and $f = 0.2$ m (0.66 ft.) travel, the impact factor is $e = 4$, whereas experience has also shown $e = 4$ with travel of $f = 0.1$ m. From these two figures follows the sinking speed $w = 2.2$ m/s (7.2 ft./sec.) with the "more correct" figure $k = 0.6$.

Intimately bound up with the loading conditions is the selection of the structural material and its permissible stresses. As a result of careful selection, possible here because of its small thickness, normal stresses of 250 kg/cm² (3,556 lb./sq.in.) for ash, walnut, and hickory, and 150 kg/cm² (2,134 lb./sq.in.) for pine and oak are permissible, provided that really all stresses are taken into account, that the wood is dry and straight-grained, that no fibers are cut, and that it is protected against moisture. As safe column strength, one may figure with an elasticity modulus of 130,000 kg/cm² (1,849,055 lb./sq.in.) for best hard wood, and of 90,000 kg/cm² (1,280,120 lb./sq.in.) for best soft wood, so that a safety factor of $S = 3$ is amply sufficient.

For material of accurately definable property the permissible stresses can be chosen so much higher as the breaking and notch impact strength, elasticity limit, and elongation at rupture are greater. In any case, the permissible stress must remain below the elasticity limit, to prevent the material from gradually becoming brittle and unreliable under repeated stresses.

Admittedly, the permissible stress may reach the elastic limit so much closer as the breaking strength is above this limit and the greater the elongation at rupture is. In case of buckling, a harder material with high elastic limit is to be given preference. As permissible stress, 2/3 of that at which the material is permanently distorted, is recommended.

The strength of the complete airplane can be proved by a load test, the airplane being turned on its back and the wings loaded with sandbags. Now certain safety factors are necessary according to the excess loading with sand in relation to the weight which, while expedient are, however, not altogether logical. For estimating the safety factor in the technical sense, obviously all imaginable loadings should be included. The worthwhile safety factor is that at which all stresses remain below the elastic limit, i.e., at which no permanent deformations remain after unloading. This demand is in no airplane construction complied with, because all use fittings which gradually stretch as, for instance, cable eyes, bolt holes in wood, steel cable, fabric covering, etc. A load test with actual service load for the purpose of using the amount of permanent deformation as a measure of the quality, as is customary in bridge design, is not feasible here. Thus there remains only a load test with a multiple of the service load, which is best carried to failure, because it cannot be used any longer after such a test. The destructive load test gives certain indications of how great, assuming a satisfactory life span of the airplane, the service load may be. Advance in technique to a point where it is possible to design a structure without permanent deflection under service load, should also enable us to effect more economical load-test methods on many airplanes without damaging them (reference 33).

In the discussion following this report, Baumann claimed to have obtained a record of 2.5 times the acceleration due to gravity with a small accelerometer of his own design in a pull-out from a glide with power on. He held it doubtful that pressure strikes the wings from above. Fast airplanes and such with small angle of attack yield upon recovery a greater increment in loads than others and should accordingly be computed with correspondingly higher safety factor. A gradation of 8 to 12 times safety was therefore advisable.

Bendemann said that the analysis of the maximum wing stress which can occur while flattening out from a glide with the desired speed, should proceed from the fact that unexpected disturbances sometimes induce the pilot to effect irrationally abrupt control motions. He recommended as basis the maximum force which can be produced by a sudden pull-out into horizontal attitude.

Von Parseval: "The demand to permit a wing loading up to 5.5 times its weight seems far-fetched. It is not very often that the assumed unfavorable factors occur simultaneously at their highest amount, and for extreme cases of that kind no airplane can be built. The maximum wing load in flight follows from formula $p = 0.075\ v^2$, where v is the possibly occurring greatest endurance-flight speed." (This speed would correspond to maximum horizontal flight speed at ground level.)

Reissner evidently interpreted this reference such that he introduced the diving speed and thus obtained yet higher stresses than those objected to by Von Parseval.

Barkhausen: "Loading tests to failure have, despite much expenditure, failed to reveal much information about the nature of structures, because the break occurs at the weakest spot, which in many cases could have been detected beforehand. It requires stress-strain measurements on many members simultaneously to specifically reveal whether the actual effect of the structure corresponds with that assumed in the analysis."

This report and the subsequent discussion even at that time touched upon practically all loading conditions recurring in later years. Especially worthy of note are the demands for high safety. Reissner was in favor of breaking load factors of from 8 to 12 for tension members, and of from 12 to 18 for compression members; Baumann likewise, for from 8 to 12; Hirth for factors above 10. Therefore, the ratio of maximum speed in level flight to minimum floating speed was as a rule < 2 for the airplanes of that time. These safeguards underwent, as Von Parseval predicted, considerable modification when airplanes were subsequently used for military purposes.

Occasioned by the above-described theoretical and experimental investigations relative to the normal airplane accelerations, as well as by the wing-stress tests with sand loads, it became accepted practice then to prescribe a constant load factor in the stress analysis for new airplanes, although all formulas given for the load factors (4) and (7) contain the flying speed. The freedom of the stress analysis from the attainable flight speed, while quite expedient, is, however, objectionable unless the speed of the new airplane type really is not higher than that of the old type serving as model.

The systematic research of stress and strength of airplanes advocated by Reissner, was started in 1913-14 by W. Hoff, in the D.V.L. Unfortunately, the results of his labors were not published until later, because of the war (reference 30).

In July 1913, the W.G.L. sponsored a contest for an "airplane accelerometer" which was to record normal accelerations at right angles to the wing chord, and which was to be used for investigating the stresses of airplanes in free flight. However, the contest could not be finished because of the war. Accelerometers were not much in use in Germany, although Searle's photographic recording accelerometer was used successfully in England (reference 31).

Hoff used Bendemann's force-metering box sketched in figures 7 and 8. The fear about mounting instruments directly on the main supporting members of a wing led to the design of a cable tensometer, shown in figures 7 and 8, developed from the tensometer of Lenoir and Pocton. The test cable under stress S was carried in a light bend over three stirrups, producing in the middle the force $P = 2\,S \sin \gamma$. If γ is very small the cable pull becomes, according to figure 7,

$$S \sim \frac{P\,l}{4\,(a + s)}$$

The first experiments with this instrument were carried out in February 1914 on an Albatros-Taube which, because its landing gear served at the same time as lower cabane for the wing suspension, was particularly suitable. Two tensometers were fitted to the cables between the landing gear. (See fig. 9.) There being no danger of exceeding the elastic limit, it was permissible to deduce from the stress reversal of one wing part to one of the same kind in the wing itself. On the quite sluggish airplane the following load factors were recorded:

Climb	n	= 1.07
Gliding flight		0.94
Lateral turns		1.4
Flattening out from a glide		1.6

In July 1914, the Albatros B II, one of the oldest types of airplanes still flying, was tested in the same manner (fig. 10). Each half-wing had five test stations in the lift wires of the center section, in the wires leading from the rear upper spar to the engine as well as in the cross wires of the center section (fig. 11). The wing area was 41.7 m² (449 sq.ft.), the gross weight during the tests, 830 to 1,000 kg (1,830 to 2,205 lb.), and the wing weight, 178 kg (392 lb.). The speed, although not exactly measured, was around 110 km/h (68 mi./hr.). The cable forces are tabulated in table V. The highest load factor by pull-out from a glide is

$$n = \frac{1652}{821} = 2.01$$

as compared to level flight.

Table V. Loads on Albatros B II (in kg)

	Left wing					Right wing					Total normal force
Cable No.	d_5	d_7	f_5	f_7	f_9	d_5	d_7	f_5	f_7	f_9	
Quota of normal component	0.67		-0.56		0.46	0.67		-0.56		0.46	kg
Level flight	238	398	139	83	258	275	424	174	103	195	821
Left turn	269	409	112	118	251	261	442	154	95	203	866
Right turn	269	415	150	85	266	269	442	195	109	197	846
Corkscrew, left	482	688	-	-	336	446	738	-	-	266	-
Leveling out from turn	403	658	52	23	355	417	660	84	21	264	1554
Glide	208	354	164	98	251 204	237	374	200	109	197 148	672 628
Recovery from glide	400	700	80	23	340	453	703	120	12	252	1652
Landing impact	-	-	248	282	-	-	-	304	315	-	-643

Several indicator records are shown in figure 12.

These measurements were of great significance for the development of the German airplanes. It had been proved, even if only on a low-powered airplane, that loading condition A (pull-out) produces the highest load factor $n \sim 2$.

Wing tests had been carried on by Hoff since 1913, but the results were not published until 1916 (reference 32). The method of wing stress analysis was already briefly described in the first yearbook of the D.V.L., namely: wing-stress tests were carried to failure because many details of the airplane were not yet sufficiently developed and not in harmony with the whole. Once this stage of development has been overcome, we shall be able to estimate from the destructive tests of individual parts and from the behavior of a wing under moderate loading, whether the mathematical assumptions hold good.

This stage of development was reached about 1928 (reference 33).

In static tests the reversed wing is loaded with dry sand conformably to the magnitude and distribution of the forces. The deformations are photographed or written on vertical planes as shown in figure 13. At the first sign of failure the wing must be shored up, so that the cause of failure may be ascertained without incurring further damage. After the damage has been repaired the loading is continued. In this fashion it was possible to detect up to five weak spots in succession and to effect a marked increase in strength by a slight increase in weight. On the premise that the wings support themselves in flight the breaking strength of the wing is

$$S = \frac{Q + G_F}{n\ (G - G_F)} \tag{11}$$

The load factor n depends upon the qualities of the airplane, of the pilot, and the weather conditions but defies for the time being any reliable estimation and can therefore not be used as criterion of the structural safety. It appears much more simple to put the load factor $n = 1$ and thereby select the known stress in undisturbed level flight as measure of the breaking stress.

The thus defined safety factor is identical with the breaking load factor n_{Br} and does not represent a safety factor in the usual sense.

The first test wing of a monoplane (fig. 13) broke under 1,344 kg (2,963 lb.) total load by buckling of the rear spar. The wing was to support 400 kg (880 lb.) in undisturbed flight. The breaking load factor was $n_{Br} \sim 3.8$ and did not come up to the demanded requirements. Then the right wing spar was suitably strengthened, after which the wing carried a total load of 2,975 kg (6,559 lb.) without breaking.

After this experience a third wing of larger dimensions was designed. (See fig. 14.) The break occurred under 3,000 kg (6,614 lb.) load by buckling of rear spar near the fuselage. A subsequent check yielded for the ash rear spar a flexural strength of 797 kg/cm² (11,336 lb./sq.in.) with a combined compression strength of 134 kg/cm² (1,906 lb./sq.in.); but the actual stress might have been lower as a result of inconclusive assumptions about the spar mounting. The breaking load factor was

$$n_{Br} = \frac{3000}{500 - 67.5} = 6.95$$

Figure 15 shows the deflections of the leading edge of the wing at the different load stages of 400 kg each.

The results of these experiments, begun in 1912, in Döberitz, and subsequently transferred to Adlershof, were the "special airplane requirements" in 1913, demanded by the military authorities.

Apart from general regulations, they contained the following strength specifications: Until further notice, all airplanes shall have a factor of safety of 5 against pressure from below; airplanes with a speed of more than 120 km/h (75 mi./hr.) a factor of safety of 6. This on the basis of sand loading evenly distributed over the wings. The breaking load is figured from the loading and the wing weight. The military load for all airplanes consists of: structural weight, water, fuel, and oil, and 200 kg (440 lb.) useful load.

The military authorities shall from time to time select certain airplanes from among a series of delivered or ordered airplanes for static breaking tests.

A salient feature is the grading of the ultimate load factor according to the speed obtainable in horizontal flight. The quotient

$$S = \frac{Q + G_F}{G} \tag{12}$$

was called "safety." Hoff, in March 1914, suggested that the safety or the ultimate load factor would be more correct if computed according to equation (13), and obtained with due allowance for mean wing weights, which correspond to the "special requirements," a breaking load factor of 5.75 for monoplanes and of 6.41 for biplanes. The airplane manufacturers countered this by saying that airplanes with safety factor of around 3 had taken care of any emergency and that a sudden raise in safety by double would result in excessively heavy wings and they advocated the breaking load factor of 5 in conformity with equation (13).

4. Loading Conditions During the War

At the beginning of 1915, that is, shortly after the beginning of the war, the Inspection Branch of the Army Air Corps issued its "Specifications for the Design and Delivery of Military Aircraft," which contain the following strength requirements: A safety factor of 6 is required for the strength of the wings against pressure from below - this on the basis of a sand loading which corresponds to the air pressure distribution over the wings. The safety factor S is computed according to formula

$$S = \frac{Q + G_F}{G - G_F} \tag{13}$$

The wing weight includes the bracing wires, etc. The military weight constitutes net airplane weight, water, fuel, and oil, and the momentary useful load.

For breaking test the airplane is so supported at the points for engine, fuel, and seats that the force of gravity on the wings is at the same angle as the resultant of the wind forces in flight. The fuselage must have sufficient strength against flexure and torsion (triangular cross sections prohibited).

The engine mounting must be solidly connected with the continuous longerons of the fuselage (hull). For static engines the landing gear must be strengthened. Wings and tail surfaces must, basically, be independent of landing gear and tail skid. Special care must be taken toward solid suspension of wings, tail surfaces, and their bracing to the fuselage. The main lift wires must be strong enough so that none will break first in load tests. The steel turnbuckles must have a strength of one third greater than that of the wires.

By "safety factor" is again meant the breaking load factor. The 1915 specifications, quite inadequate for modern conception, were soon amended after publication, as follows:

The strength against failure from below requires a safety factor of 4.5; in multi-engine airplanes, a safety factor of 5. This specification was based upon erroneous conceptions about stresses in large airplanes because of lack of experience.

W. Hoff, in his report on the strength of German airplanes (reference 34) published in 1922, voiced himself as follows:

"In every branch of science the stress analysis of the structural parts is preceded by investigations into the maximum service loads. Its results are correlated with a safety factor depending upon the type and expectancy of the load, so as to ascertain the breaking strength of the structural part. The safety factor is chosen so high that the elastic limit is in no service attitude exceeded, thus avoiding permanent deformations at all times. The maximum loading of an airplane defies reliable estimate and can only be judged on the basis of comparisons. The 1914 measurements on the Alb B II have shown that a static load twice as great as the service load can occur. In light and easily handled airplanes (pursuit), this figure should be higher; in multi-engine airplanes, lower. These service multiples of the static load should, strictly speaking, be multiplied by a satisfactory safety factor. Then the multiple of the static load is obtained which is decisive for the analysis of the breaking strength. In airplane design the method of first choosing the service multiple by one figure and then the safety factor by a second figure, has the great disadvantage that an agreement between selected figures must be effected twice. It was therefore decided to agree to the sum of the two figures, but to leave the question, into what factors these products could be divided, open. Unpropitiously in airplane design, the product is often called safety factor, and must be guarded against, because it may create false conceptions about the actual structural strength of an airplane. The following consideration at that time yielded the 4.5 times breaking load:

"The elastic limits of the principal materials used in airplane construction - wood and steel - diverge very considerably from the breaking strength of these materials. For curved wood (wing spars), figures much less than 50 percent of the breaking strength are in order. For steel the figures are higher, depending on its hardness. H. Dorner and E. Heller, who were responsible for the strength of airplanes, suggested to assume the elastic limit for wood at about 45 percent of its breaking strength or, computed with the scale of the load factors, at twice the load factor. They inferred that then the breaking strength would be $2\,\frac{100}{45} = \sim 4.5$ times the load. This is also applicable when wood of less than 45 per cent breaking strength is used, since the stress in a spar in buckling and bending is not proportional but increases at a higher rate than the loads, so that even by double the load factor it is still less than 45 percent of the breaking stress and, consequently, below the elastic limit."

Reissner and Schwerin published a comprehensive report on the stress analysis of airplane spars, in 1916 (reference 35).

The writers fail to see in literature a stress analysis on airplane wings in accord with the modern methods of statics, such as is used in bridge design.

There are three kinds of tension members in an airplane (biplane) wing: 1) spars or flanges, 2) uprights, and 3) oblique members (wires). The first two are stressed in bending and buckling and therefore dimensioned as to be little subjected to length changes. But the oblique wires stressed only in tension are very elastic and of great strength, hence subject to much greater length changes. The results are marked angular changes in all triangles of the system and through it considerable bending moments of the spars, which, owing to their additive load by the lift forces and their restricted height, are subject to considerable deflections. Through the column stress of the spars the proportionality existing in other trusses between loads and stresses is lost, so that the question of actual factor of safety of a wing demands a separate analysis.

The loading conditions should be as elementary and comprehensive as consistent with the safety and light weight of the system.

The spar loads for the polars in figure 3 are computed conformably to equations (3) to (5), because they mathematically confirm the practical theories about the great load on the rear spar from below and on the front spar from above, and do not seem to be appreciated enough. (See fig. 16.) The curves reveal a marked downward load on the front spar ($A > 0.875$ G) in diving and an upward load ($B > 0.75$ G) on the rear spar. Then follows the static calculation of a lift truss on a biplane and two monoplanes. First, the principal stresses of the lift truss are computed for hinged joints and the continuous spars as bending resistant beams with stated support deflection with due regard to column effect by the principal stresses. The analysis is carried out with an assumedly constant modulus of elasticity (Hooke's law) for load factors 1 and 3, and shows the bays in which the stresses as result of column effect increase faster than the loads. These and other similar calculations were submitted to the D.V.L. The Inspection Section later demanded the stress analysis in proximity of the breaking load, although not only the buckling limit lies in this range but it is also no longer possible to speak of constant elasticity modulus. Reissner and Schwerin consistently advised against this request, but the authorities attached some significance to these fictitious calculations because they believed to gain from the comparison with the actual breaking test, some points for refinement of the design specifications.

However, the agreement of these "fictitious calculations" with breaking tests was often remarkably close, so that even to-day such breaking-load calculations are much in favor in airplane design practice.

The 2d edition of the BLV, in 1916, attempts to conform to the above outlines of Reissner and Schwerin, relative to the loading conditions. The development of the loading conditions and of the airplane analysis has been in the hands of Hoff, Madelung, and van Gries since 1916.

The majority of the German military airplanes was designed according to the 1916 BLV specifications, which are repeated here verbatim.

The 1916 BLV Specifications

A wing strength which is a multiple of the loads in level flight, is demanded against air loads in pull-out, glide, at the nose and from above. The gross weight by full load, i.e., with load empty plus momentary useful load, shall be introduced.

Figure 17 pictures the directions of the air loads and table VI, the required multiple.

Table VI. Prescribed Strength against Air Loads (Structural Safety)

L o a d Case		E and D types	C and G types	R type
Pull-out	load A	5.00	4.50	4.00
Glide	" B	3.50	3.00	2.50
Frontal pressure	" C	2.50	2.00	1.50
Excess pressure	" D	3.00	2.50	2.00

The strength against air loads (structural safety) shall be proved by static analysis.

If necessary the wings shall be tested with sand loads equivalent to the air loads. The safety factor is computed according to formula (13).

Landing gears shall not form any inside part of the lift truss. All parts which are easily damaged, such as turnbuckles, struts, and attachment fittings shall have an excess breaking strength of at least 200 kg (440 lb.). The tail skid must form a separate unit.

Prescribed breaking strength of movable and fixed surfaces:

1. Fixed surfaces (alone, without movable surfaces) 300 kg/m²

2. Fixed surfaces attached to movable (fixed surfaces not loaded) 150 kg/m²

3. Balanced movable surfaces not attached to fixed surfaces: E and D airplanes 200 kg/m²
C, G, and R airplanes 300 "
(kg/m² × .204818 = lb./sq.ft.)

The strength of all parts of the controls must correspond to these movable surface loads. In addition, hand controls shall correspond to a breaking load of 80 kg (17.6 lb.) and foot pedals to a divided breaking load of 300 kg (66.1 lb.) (eccentric for hand wheels).

The controls shall be rigid enough to insure ample clearance for operation even by breaking load of elevators. The control friction by breaking load shall not exceed one fifth of the control force.

The fuselage shall be sufficiently rigid in bending and torsion, especially in the region of the cockpits.

The engine longerons must be so placed that shocks are immediately transferred to the whole engine and deformations of the fuselage (hull) do not affect the engine.

The main fuel tank is to be solidly anchored. To protect the passengers when nosing over or capsizing the mounting from the front and upward must be able to sustain twenty times the weight of the full fuel tank.

The landing-gear struts and, if no buckling supports are provided, the strut sockets at the fuselage shall be detachable. The height of the landing gear is prescribed as follows:

With a tractor propeller, when the plane of the wing (measured near fuselage) is level, the ground clearance must at least be 20 centimeters (7.87 inches). The same applies to pusher propellers with tail skid on the ground.

The shock absorption must be at least 0.21 times the airplane weight (mkg) and 0.32 times in the absence of tires. The elastic travel must be provided with a stop.

The tail skid while elastic and movable sidewise, must be stable. Its absorption must be at least one eighth of that of the landing gear.

All wires, cables, etc., especially the eyes, must be tested to 0.5 breaking load before installation.

Adjustable safety belts must be provided for every passenger. The belt must be 15 cm (5.91 in.) wide and be able to hold the body by a pull of 300 kg by 10 cm stretch. The attachment to the fuselage must have the same strength.

Static Analysis

1. The cross distribution of the air loads and masses across the span of upper and lower wing assumed according to the design drawings and the weight analysis. Comparative analysis with various assumptions of lift decrease at wing tip, distribution of loads over the panel points.

2. Analysis and stress diagrams of main and inside wing truss.

3. Proof of strength and safety in buckling of spars and wing tubes, struts, and internal members, wires, turnbuckles, fittings in comprehensive tables with data on material strength, elasticity, elongation, lengths, sectional areas, inertia, drag moments, and on the forces, moments, stresses, column and breaking strength of each member.

By column and breaking strength is meant the ratio of the existing to the required strength.

The analysis shall be made for the four loading conditions and that of the control system, with the given loads.

The stresses of spars in bending and buckling shall be computed according to Hütte, vol. I, strength of straight members, or according to formula

$$M = M_o \frac{S}{S - 1} \tag{14}$$

where M_o = moment without buckling, S = column strength, or else according to Müller-Breslau's "Graphic Statics," vol. II, no. 2. Proof must be given that the stress of a strut bent through 1/200 of its length under impact, under full load does not exceed the elastic limit.

Short members stressed in buckling shall be computed according to Tetmajer's formula. The minimum strength as defined by test is to be used as material strength. The special material is used, test samples shall be submitted which admit of comparable measurements.

Now that a mathematical proof of the strength was prescribed, it was possible to consider several load cases. Before that time one breaking test sufficed (for case A), because the destruction of more airplanes for further load tests would have been prohibitive. The c.p. displacements had to be disregarded because of lack of sufficient wind-tunnel data, and the mean values used instead were not too propitious. The A case about corresponds to the attitude by maximum lift coefficient $c_{a_A} \sim 1.2$, the B case to gliding by $c_{a_B} \sim 0.3$. The minimum resultant of the air load in a dive of airplanes at that time being estimated at $c_{r\ min} \sim 0.10$ to 0.17, the breaking load factor for case B is

$$n_B = \frac{c_{a_B}}{c_{r\ min}} \sim 6 \text{ to } 10\ c_{a_B} = 1.8 \text{ to } 3.0 \tag{15}$$

an assurance for the strength of the wings by pull-out at high speed in the B-case condition, so long as the maximum acceleration to which the pilot is accustomed is not exceeded.

Another explanation of case B is, that in the reliable airplanes at that time the front and rear spar were of about equal strength for structural reasons. (See fig. 2.) Owing to the c.p. displacement, the breaking loads of such a wing in case A and case B were approximately as 3:2, as proved by Reissner in his report, cited above. Hence the next step was to retain the ratio of breaking-load factor $n_B:n_A = 2:3$.

Whereas, aerodynamically, the B case is justified only by the first-quoted consideration, one finds later in all loading conditions a constant ratio of the load factors

$$n_B = 0.6 \text{ to } 0.8\ n_A,$$

which, for acrobatic and other airplanes with high case-A load factor, leads to an unnecessarily great and no longer usable case-B strength, when for c_{a_B} the figure 0.3 or the often still lower lift coefficient for maximum level flight is used.

The D.V.L. in 1927 measured $c_{r\ min} = 0.115$ in a dive on an acrobatic airplane. So the possible load factor is $n_B \sim q\ c_{a_B} = 2.7$, whereas a 7.2 breaking-load factor is demanded. R. Voigt (reference 36) put the load factor at $n = 10\ c_a$, and likewise arrives at the result that the well-known loading conditions for pursuit airplanes require a too high case-B strength. This assumption is also unfavorable for commercial airplanes because they scarcely have a speed at which the strength of case B is fully utilized. For modern wings with high torsional stiffness or wing sections with fixed c.p., the case B has, moreover, lost in significance and its importance now is restricted to the classic two-spar type of wings.

A notable feature is the oblique direction of the force in the case-B loading, which was determined knowingly from the wind-tunnel averages in order to insure ample rigidity of the incidence bracing.

No account was taken of the attainable diving speed and the wing polars in the C case, but a breaking torque of the wings of at least 1.0 to 1.67 $(G - G_F)$ t to be taken up by the fuselage was demanded, which was to take care of the most unfavorable cases. By assuming a moment coefficient $c_m = 0.1$ and a safety factor of 2 for diving, the minimum coefficients of the resultants of the air loads given previously, are obtained.

Owing to the absence of numerical stability data in the BLV specifications, the summary establishment of breaking loads per unit area of tail surface was the cause of the tendency toward smaller control surfaces with less span which, aerodynamically, were not effective enough.

The landing gear was to sustain a landing at 2.5 m/s (8.2 ft./sec.) sinking speed on level ground, and the energy absorption was to be accordingly. Sufficient travel to lower the stress when passing over obstacles was not provided for until later.

With the aim of making the regulations as simple as possible, the tests were all static tests, no dynamic test of the forces in the whole airplane being demanded. Instead it was assumed that, for example, the fuselage was restrained at appropriate points analogous to the load tests. Although the 1916 BLV did not make special mention of it, one generally figured that in biplanes the upper wing had to carry 1.1 times, and the lower wing, 0.9 times the wing loading.

Based upon war experiences (Germany alone built more than 40,000 airplanes) the BLV regulations were revised in 1918, by Hoff, Madelung, and Stelmachowski. The strength specifications were separated from the design specifications and apponded as "Guiding Principles for Strength Proof of Airplanes."

The 1918 edition revealed many modifications, although the fundamental principles had been retained. Because of the insistent demand for climb and ceiling, which was limited by the power of the available engine types, the number of design types did not exceed 100, but they were built in large series and a world of experience was accumulated in a short time. To-day the number of design types is greater, the output per type smaller, and the selection of the right loading condition more difficult.

Design and Delivery Specifications (BLV), 1918

Main load cases for wings

The analysis shall be made for four main load cases, conformably to the different flight attitudes. Direction of loads and point of application are shown in figure 18. The minimum theoretical breaking load is the multiple of total weight minus wing weight, given in table VII.

Airplanes falling into categories I and II are exempt from inverted flight test (D case).

The load factors given in table VII apply only for mathematical proof when the plate effect of the covering of the edge and intermediate strips and of the ribs on the spars is not allowed for.

These propitious effects gain full validity in the strength test. In deference to that the strength test is governed by the load factors appended to table VIII.

Table VII. Applied Load Factors

No.	Category Status 1918 edition BLV	Applied load factor A case pull-out	B case glide	C case dive	D case inverted flight
	Airplanes with				
I	gross weight over 5,000 kg	3.5	2.5	1.2	-
II	gross weight exceeding 2,500 to 5,000 kg (useful load 1,000 to 2,000 kg)	4.0	2.5	1.5	-
III	gross weight 2,500 to 4,000 kg (useful load 800 to 1,500 kg)	4.5	3.0	1.75	2.5
IV	gross weight 1,200 to 2,500 kg (useful load 400 to 800 kg)	4.5	3.0	2.0	2.5
V	gross weight up to 1,200 kg (useful load up to 400 kg)	5.0	3.5	2.0	3.0

(kg × 2.20462 = lb.)

Table VIII. Load Factor Specified for Strength Test

Airplane types	Specified load factors A case pull-out	B case glide	C case dive	D case inverted flight
I	4.0	2.5	1.2	-
II	4.8	2.6	1.5	-
III	5.5	3.2	1.75	2.8
IV	5.8	3.3	2.0	2.8
V	6.5	4.0	2.0	2.5

The military authorities reserve the right to reject the existence of the strength proof in the case that the stipulated load factors are precisely obtained, but at which deformations occurred which in accordance to their own experience prove an unsuitable design.

The diagram of the air loads, given in figures 18-22, is reserved for spar, strut, and cable analysis. Details regarding the loading and analysis of ribs will be found in T.B. (Technische Berichte), vol. I, p. 81, and figure 23.

In the C case, the loading consists of frontal pressures C_o and C_u equal to the resultants of the air loads acting as upward pressure on the rear portion of the wing, those acting as down pressure on the fore part of the wing (fig. 21) and of the turning moments M_o and M_u.

The load factors in the C case apply to frontal pressure C_o and C_u only. The total load on the wing is computed as $C = C_o + C_u$.

For wing-truss analysis without a multiple the moments shall be

$$M_o = C_a \times 1.75\ t_o \quad \text{for upper wing}$$

$$M_u = C_u \times 1.75\ t_u \quad \text{for lower wing.}$$

In rib investigations these moments shall be 50 percent higher.

If the wings have decalage, one experiences lift, the other, drag. Lift and drag are inversely equivalent and are obtained either from the polar diagram or else shall be estimated at 1 G for every decalage, and evenly distributed over both spars.

The next chapter treats of the relation of load absorption of upper and lower wing of a biplane with different stagger and wing incidence based on wind tunnel tests for the four load cases.

Decrease of Lift at Wing Tips

At the wing tips the load per unit length p, otherwise evenly distributed across the span, drops to p/2 (fig. 24) over a distance equivalent to the mean rib depth.

This assumption is valid for investigating the inner bays of the spars only. When computing the overhang, the full load p, effective up to the wing tip, shall be introduced.

The load per unit length p stresses the wing cellule in two ways:

a) by producing longitudinal forces in the members of the cellule and

b) flexural stresses in the spars.

The flexural stresses set up by the partial forces of p, which fall in the plane of the chord, can be ignored as soon as the wing because of ribs, internal bracing or strutting, acts as homogeneous plate.

Panel Point Loads

These loads are deduced for the four load cases for which the air loads and the loads per unit length have been determined. The most elementary assumption is that half of the transverse loading of the bays is transferred to the joints.

Improvement of these panel-point loads according to the calculation of the bearing moments from the elementary Clapeyron equations, is readily effected with due regard to any existing displacements. Determination of the ultimate panel-point loads from the general Clapeyron equations is desired.

Loading of Tail Surfaces

The mean loading per unit area of fixed and movable tail surfaces is to be effected in accord with

Table IX. Specified Mean Tail Loading

Airplane type	I	II	III	IV	V
Mean loading kg/m^2	120	120	150	180	200

The aileron loading q is effected at 200 kg/m^2.

The effect of unsymmetrical wing loading, as in sharp turns, for instance, as well as the influence of an unsymmetrical mass effect by oblique landing, especially on cabane, center-plane section, and center of fuselage, is to be investigated.

In monoplanes, unbraced biplanes, and such with one plane of bracing, the strength test shall also adduce that the warping between the spars, measured at the wing tips, is no more than 5° in the A case, and no more than 10° in the C case.

Strength Factors of Materials

A final proof of the strength built up on generally known averages is not permitted.

As concerns materials with accurately known strength factors, these factors, as well as the other material qualities, such as elongation, (Young's modulus, etc.), shall be determined by test and the obtained minima used in the analysis.

For cables and wires, the elongation law must be proved in each individual case by test on at least three full-sized samples, with due regard to thimbles and splices.

As to spars, the raw material quality figures must be proved in each case by tests conforming approximately to the actual loading attitude.

Breaking Strength - Basis of Strength Proof

For air loads on the airplane maximum values are chosen so that the wing stresses computed therefrom may approach the breaking limit. The selection of these maxima is on the basis of the reasoning that, first the breaking stress of the most used material - wood - is readily attained from test specimens, whereas the elastic limit fluctuates; second, there is no simple relationship between loading and stressing a member in bending and buckling; for example, in such a case the bending moments under fourfold load are greater than double the moments under twice the load.

Cross Bracings

These shall be analyzed on the basis of the load by mass effect while landing (six times wing weight), provided that none of the principal load cases A - D makes greater dimensions necessary.

When the landing wires do not conform to one of the main load cases, their cross sections shall be 70 percent of that of their corresponding main wires, even if the test for mass effect in landing would permit of smaller sizes.

Struts, Wires, Cables, Turnbuckles, Fittings

Unless covered, these parts shall have a strength margin of 200 kilograms each.

In struts the effect of the initial stress is frequently higher than that of the air loads. It must be proved that the longitudinal force in a strut is below Euler's buckling load, and that by an initial deflection of 1/200 strut length the occurring stress under the effect of 1/2 the specified breaking load does not exceed 50 percent of the breaking strength.

Short members stressed in compression, if the slenderness ratio $l:i \leq 105$ for ingot iron and steel and < 110 for wood, shall be computed according to Tetmajer's column formula.

Fittings, sockets, connections, and turnbuckles must always be stronger than the wires. Instead of their analysis, official strength test reports may be submitted and the description must be such that the test can be repeated if deemed necessary. The fittings, in particular, must be tested very carefully, because the strength of the whole cellule is endangered by a weak fitting.

Fuselage

The stress in the longerons and diagonals shall be analyzed from the loading acting simultaneously on horizontal and vertical tail surfaces. Besides, ample strength in compression and buckling of members in the region of the cabane and of the body part between the wings must be proved under six times the fuselage weight from above (nosing over). When picture-frame cabanes are used which, by equal dimensions, are less rigid compared to such diagonally braced, the effect of their deformation must be shown in the analysis of the statically indeterminate quantities of the wing cellule.

The load on the hand control shall be assumed at 80 kg; the foot control must be able to maintain a load of 300 kg, 150 kg on each pedal.

For the suspension of the gasoline tank with useful load and superstructures, the force applied horizontally and vertically is assumed at 8 times the weight of the full tank for categories I and II, 15 times for category III, and 20 times for the airplanes in categories IV and V.

The loading of the passenger seats shall be, in view of the mass effect:

at least	200 kg	for categories	I and II,	
" "	300 "	" "	III " IV,	
" "	400 "	" category	V.	

Fins and Rudders

In calculations of fins (fixed), rudders (movable), their bracing and fittings, the loading per unit of surface shall be raised to 300 kg/m² (for categories I and II, to 200 kg/m²). Members stressed in buckling, etc., shall be analyzed for an initial deflection of 1/200 of this length.

Aside from the calculation of the rudder surfaces themselves, the torsional stiffness of the rudder axes and the flexural strength of the levers shall be proved (applies to elevator, rudder, and ailerons).

Landing Gear

Here three loading conditions are assumed (fig. 25), namely, one-sided impact from below, from the front, and from the side. The loads A and B, as well as A and C shall be assumed as acting simultaneously. The loads shall be at least the multiple of the static wheel load as given hereunder (50 percent of airplane weight for double wheels).

Load	Multiple of static wheel load
A	6
B	4
C	0.6

The energy absorption of the landing gear is figured (in mkg) at

Gross weight (kg) × 0.18 m with automobile tires
" " " × 0.26 " tire substitute.

The specifications further require that the travel should be 10 to 15 centimeters, with a stop.

The Imperial Navy published in 1918, a set of "General Design Specifications for Seaplanes" along the lines of the 1918 edition of the BLV, but differing as to height of load factor, etc., namely:

Table X. Loading Conditions for German Seaplanes

Load factor	E class (to 2.5 tons)	C & G class (to 5 tons)	R class (over 5 tons)
A case pull-out	4.5	4.0	3.5
B " glide	3.5	3.0	2.5
C " dive	2.0	1.75	1.2
D " inverted flight	3.0	2.5	-
Static tail loading (for fuselage analysis) (kg/m²)	150	120	100
Dynamic tail stress (kg/m²)	225	225	150
Ailerons (kg/m²)	150	150	125
Gasoline tank anchorage, load factor	15	10	6

No data are given about stresses in seaway. Statically indeterminate float gears have proved most reliable. Special attention must be given to the bottom strength in the fore-and after body.

All seaplanes must be capable of being hoisted. Hoisting gear shall be analyzed with five times the breaking load factor. No distinction is made between theoretical and experimental breaking load. Note the higher load factors in the B, C, and D cases of the first two classes up to 5 tons gross weight, although the point of application of the load is the same as in figure 18, of the 1918 BLV.

Stelmachowski, who took part in the tests on military airplanes as well as in the compilation of the 1918 BLV regulations, read a report in the summer of 1918 before an audience of airplane designers, intended to explain and give reasons for the 1918 loading conditions of the BLV (reference 37).

The design, or the evaluation, of an airplane proceeds from two points of view:

1. Satisfactory flight qualities; and
2. Full development of these qualities without failure under unfavorable stress.

The exploration of the air loads on an airplane has not yet reached the stage where they yield to mathematical treatment in all cases and, where an apparently clear picture of the effect of the air loads is available; we lack the mathematical tools to express the phenomena in a practical form. Neither do all phenomena lend themselves to solution by experiment or measurement in free flight, for such maneuvers as a pilot, in moments of danger, attempts or executes instinctively, cannot be emulated at will. And these are the very cases in which the airplane is most frequently stressed to the danger point.

Consequently, we must chiefly rely on experience, as far as analysis of the loadings is concerned. The usual procedure is to assume that one airplane of known strength has proved capable of withstanding all air loads imposed upon it, even in the most severe cases, whereas another, not quite as strong, was unequal to the task; hence a strength must be prescribed which is greater than that of the second, and not greater than that of the first airplane, Now, if the strength of different airplanes is not greatly at variance, as actually is the case, the choice is fairly well limited. The load factors not being the same in different flight attitudes, these attitudes are expressed by four "principal loading conditions or cases A to D."

It would, in fact, be erroneous to select the same load factors for these load cases for all airplane types. The air loads on the wings correspond to the accelerating forces of the airplane in motion and they depend upon the speed change, that is, they are greater as the airplane is faster and more maneuverable. Besides, since it is accepted practice to make the desired speed and maneuverability conditional upon the gross weight or the useful load, it follows of itself to classify the airplane types according to weight and useful load into different "stress categories" of which there are five at the present time.

Admittedly, no classification can do all individualities full justice, if it is to remain comprehensive. The mathematical estimate of the strength of individual members of the cellule is not identical with the proof of the strength by test loading. In the calculation the wing cellule is considered a framework consisting of individual members, whereas, as a matter of fact, the wings do not act as beams consisting of two flanges (spars) and fillets (internal braces) each, but in part as homogeneous plates:

1) because of the relatively closely spaced and spar-connected bending-resistant ribs which, when one spar is overstressed, transmit at least part of it to the other under less stress;

2) because of the unloading effect of the ribs, the edge strips and intermediate strips on the internal bracing;

3) because of the not inconsiderable contribution of the covering toward the load distribution.

These effects are disregarded in the analysis. For that reason it requires two regulations: one for mathematical proof of the strength with low, the other for test loads with high load factors in the load cases A, B, and D, but not in case C, because in view of the dimensions of the internal bracing a high frontal pressure is specified for the mathematical proof and because the plate effect of the wings in a dive is of less validity relative to the turning moments of the forces. (This consideration is rather abortive, especially with a view to the compound rib effect.)

The result of these considerations are the load factors in tables VII and VIII specified for the stress analysis and for test loading. These loadings are to be interpreted as breaking loads, i.e., under this load the members of the cellule may be stressed to within breaking strength. But that does not say that the loads obtained from the tables agree with the actual air loads. Although the magnitude of the air loads is not exactly known, it may, nevertheless, be said that no such high air loads occur. Thus the specified loadings contain a certain safety margin which should be around two times, and properly so, in view of the type of loading and the quality of the raw materials. Hereby the following effects must be borne in mind:

1. The load on the airplane in flight is not steady but changes at times very rapidly; this has a vitiating effect on the qualities, especially the strength of the structural materials with respect to time. Added to this are vibrations owing to the rapidly changing loads and to the engine vibrations.

2. The difficult or altogether unprovable bracing of the lift wires set up initial stresses in the structural members.

3. The weather in time affects the strength of the structural members, especially when wood is used for spars, struts, ribs, etc.

4. Experience proves that airplanes manufactured on a production basis are, because of subsequent strengthening and installations, usually heavier than the prototype on which the static tests had been made.

All this supplies the basis for the reasoning to interpret the loadings given in the tables as breaking loads of the original airplane for static loading. And it would therefore be misleading and almost unthinking to assume a safety margin which would let the designer assume that exceeding the strength of any structural material might be permissible.

The increments to the load factors in table VIII against those of table VII have been arrived at by experience. The comparison of the computed breaking loads with those from breaking tests revealed discrepancies up to 30 percent. However, since they decrease with increasing size of the airplane, the increments are in stages. Until further experiments prove the conventional load cases unreliable, they may be assumed as conformable to reality.

According to more recent aerodynamic researches, the turning moments of wings heretofore assumed for case C at

$$M = k\ t\ (G - G_F) \sim 0.85\ k\ c_w\ t\ F\ \frac{\rho v^2}{2} \qquad (16)$$

were too small; the coefficient should be $k = 1.75$ instead of 0.67.

But experience teaches that the moments and load factors - apart from internal bracing of the lower wings -

afforded to ample dimensions; hence it would not be justified to increase the moments. So in order to arrive at the known moments again, the load factors for case C are omitted with the moments. But to avoid unduly weak internal bracing, the partial forces acting as frontal pressures in the plane of the wing are multiplied by corresponding load factors.

This regulation contains no contradiction; it allows for the fact that the spars are suitably dimensioned even in the other load cases, whereas the internal bracing is chiefly designed for diving. On top of that, a large proportion of the air loads in a dive is taken up by the fuselage, landing gear, etc., which is not included in the stress analysis nor in the load test.

Hitherto it had been assumed that the distribution of air loads was even across the wings. But that is not exactly so, especially at the tips and the parts blanketed by the slipstream. Thus in the analysis of the center bays of the spars, a space equal to the rib chord must be figured with a decrease in evenly distributed loading p to $p/2$ at the wing tips. For the spar overhang, however, the full loading p up to the tips must be assumed.

All these considerations refer to a load symmetrically divided on both sides of the center axis. But in a turn the load ceases to be symmetrical. Still no special load case is introduced, because the asymmetrical load stresses, chiefly the cabane or center section and its supporting fuselage members, unfavorably.

When defining the air loads on the tail surfaces the same obstacles are encountered as with the wings. For the tail surfaces themselves, the prescribed loading would serve no useful purpose, because they are so small that even an abnormally high load would not entail any appreciable weight increase. The tail loading is important for the design of the fuselage. But because of its compact shape, the latter must always be considered as a whole and because the discrepancies in the moments on the fuselage are slight by different positions of the mean forces of the tail loading, only the amount of tail loading is significant. Thus the problem narrows down to finding the maximum appearing tail loading.

The tail loading can be expressed by

$$q = \frac{\rho v^2}{2} c_n \tag{17}$$

and is primarily dependent upon the speed and on the coefficient. The maximum speeds are either approximately known for a given type or else specified. The maximum of c_n depends on the stabilizer setting and on the elevator displacement, and then only the most unfavorable case needs to be considered. In this manner the loads, compiled in table IX, have been set up, and they are in close agreement with the latest experience.

However, in view of possible damage during airplane shipping the loading for members of the stabilizer in group I should be raised to 200 kg/m² and for those of groups II, III, IV, and V to 300 kg/m². Ailerons are customarily figured at 200 kg/m².

The load factors for the wing ribs in cases A, B, and D shall be the same as for the whole wing cellule; in case C, the moment specified for the cellule should be raised 50 percent. The ribs must be subjected to a test loading. The static tests must conform to established practice and, for the present at least, be carried to destruction.

In the discussion following this report, van Gries suggested for use in stress analysis instead of the breaking load, a smaller load factor occurring in flight and therefore corresponding more closely to reality, and also to use the term "safety" again. The calculation could then be made within range of the elastic limit and would be more accurate.

Hoff replied that so long as we did not know the maximum service stress of an airplane, the Government Inspection Branch preferred the conventional breaking-load factors rather than the individual factors: safety and maximum service stress.

This opinion was also voiced by Kaiser, Mann, Müller-Breslau, and Reissner at a previous meeting of leading statisticians.

Madelung spoke on the unsymmetrical stresses of the cabane caused by mass forces on the wings, when one wheel touches the ground.

When assuming an evenly distributed sand load over the tail surfaces, there is the danger that the designer may be misled to save on size of tail surfaces, as actually admitted to me from several sources. Proceeding the other way, that is, assuming that moments act on the fuselage as result of the air loads on the wings, it will be seen that these moments are independent of the size of the tail surfaces. Its size may perhaps be deduced from load case B.

Sabersky stressed the importance of strength of the structural components for, as he said, the failures always occur at the same places on our field service airplanes. If the safety factors for the important joints were raised, it might perhaps be possible to lower the now specified breaking load quite considerably.

In his 1922 report on the strength of German airplanes (reference 34), Professor Hoff had this to say on the subject of strength of tail surfaces:

"The loads on the tail surfaces expressed as product of dynamic pressure, air-load coefficient, and safety factor are purely empirical figures. It is of interest to learn the factors of this product. Wind-tunnel experiments concede that the figure $c_{a_H} = 0.7$ may be considered high for tail surfaces of conventional designs. By assuming a safety factor of around 2, the 1.4 part of the loadings of tables IX and XI would have to be introduced as mean dynamic pressure of the group, which for 0.125 kg s^2/m^4 air density would correspond to the speeds given in table XI.

Table XI. Tail Surface Loading

Group		I	II	III	IV	V
Mean breaking load	(km/m^2)	120	120	150	180	200
Coefficient	c_n	1.4	1.4	1.4	1.4	1.4
Calculated speed	(km/h)	135	135	150	165	175
Actual mean top speed	(km/h)	132	-	151	170	193
Corresponding c_n factor		1.43	-	1.36	1.29	1.12

The special emphasis on the unit surface loading of German tail surfaces was rather from the point of view of insuring propitious strength conditions than for aerodynamic reasons. The characteristically German tail surfaces, narrow and deep, are the results. Efforts to overcome this were not lacking. A suitable way appears to be to utilize the tail-surface moment, which acts contrary to the wing moment for stress analysis.

The development of these specifications can also be closely followed up in van Gries' book on Airplane Statics (reference 38).

Performances of Airplanes Built During the War

Part I is suitably closed with a review of the performances of war airplanes, since all later loading conditions were largely based upon war experience. Figures 26 to 28 show the results of the wing-loading tests for the period from November 1915, to December 1917. The load factor is plotted against the airplane gross weight. The marks o denote wing failure; the other tests, marked •, were stopped prior to failure. Figures 29 to 31 show the load tests on tail surfaces.

Figure 32 shows the wing loading of the separate groups versus wing area. Owing to the military demand for high ceiling, the wing loading $p \sim 50$ kg/m² represented the constructional limit at that time.

In figure 33 the maximum speed v_h, obtained in unaccelerated level flight, is plotted against the performance loading G/N. The average is

$$v_h \sim 121 \sqrt{\frac{N}{G}} \text{ (m/s)} \tag{18}$$

Figure 34 shows the dynamic pressure q_A, at which, in case A load, the wing reaches the breaking load, versus the dynamic pressure q_h for maximum horizontal flight. The average obtained quite frequently, is

$$q_A = 1.5 \, q_h . \tag{19}$$

Translation by J. Vanier,
National Advisory Committee
for Aeronautics.

For Part II, see N.A.C.A. Technical Memorandum No. 717, which follows.

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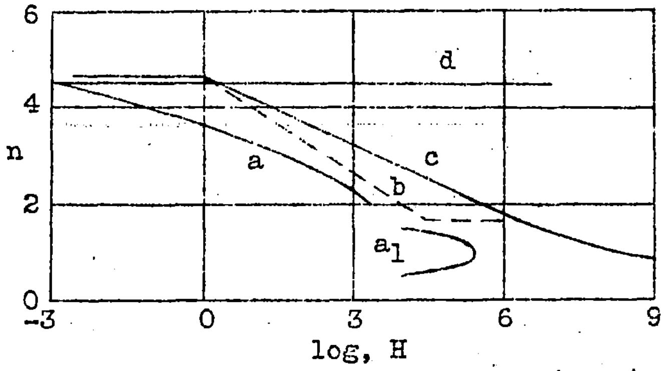


Figure 1.-Expectancy of wing stresses (a, a_1), and fatigue strength of compenent parts (b,c).

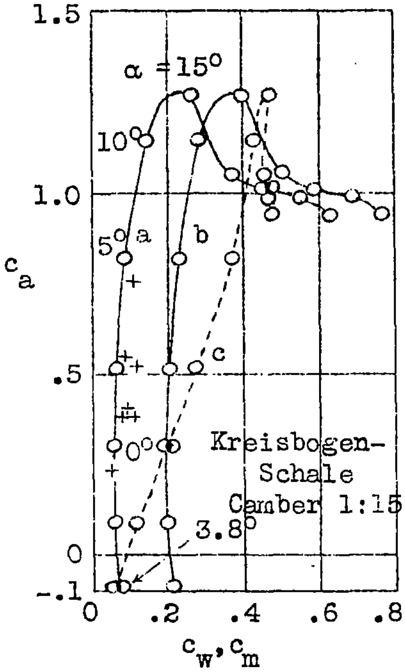


Figure 3.-Polar of a 1911 airplane according to Reissner.

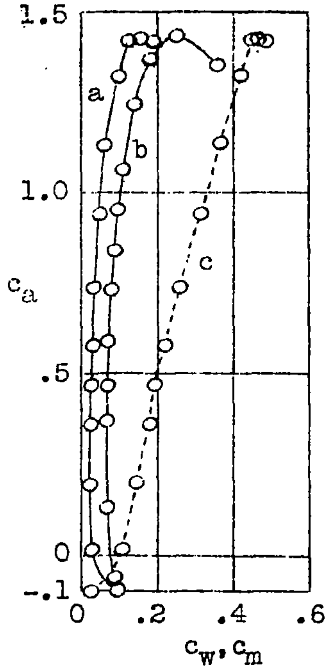


Figure 4.-Polar of a 1927 multiengine landplane. Curve a: wing polar, curve b: airplane polar curve, curve c: moment coefficient.

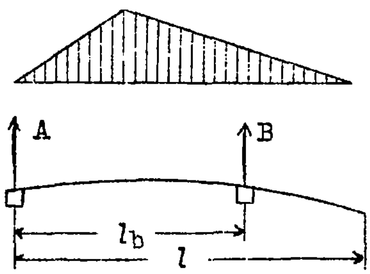


Figure 5.-Position of wing spars and rib load

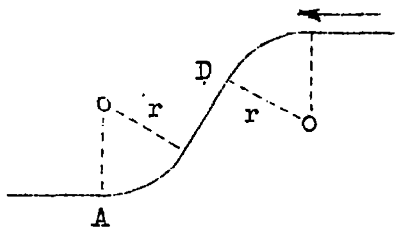


Figure 6.-Change to steep glide and pull out in horizontal flight.

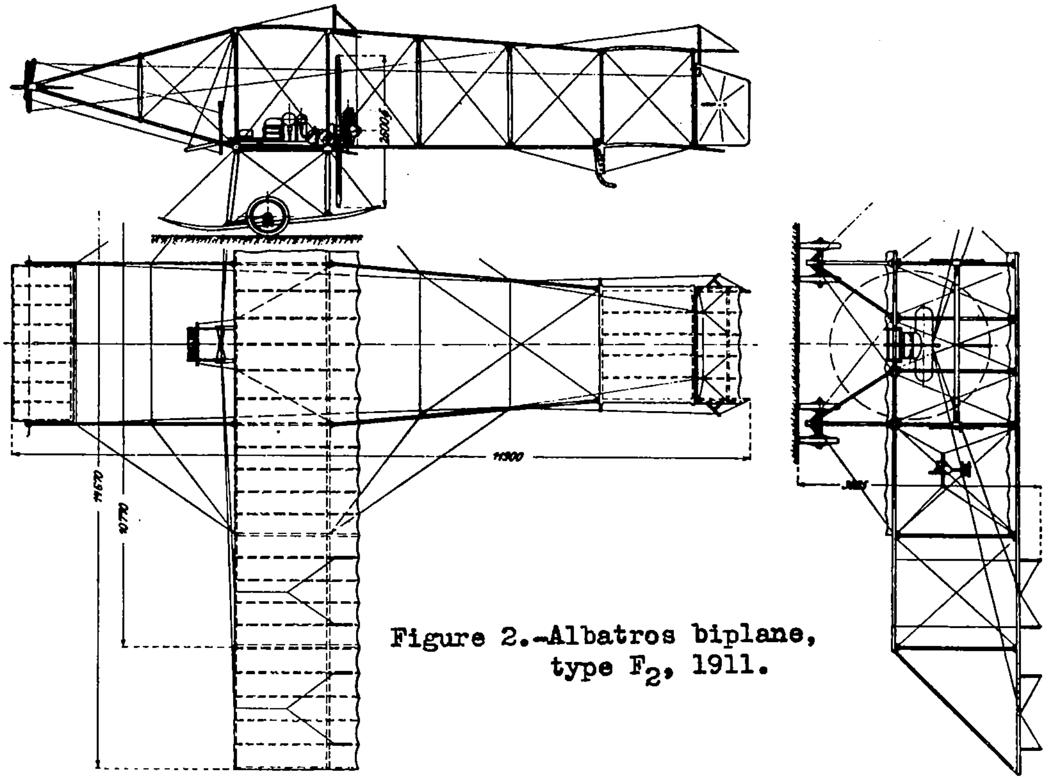


Figure 2.-Albatros biplane, type F_2, 1911.

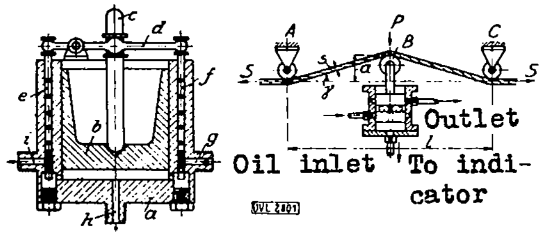
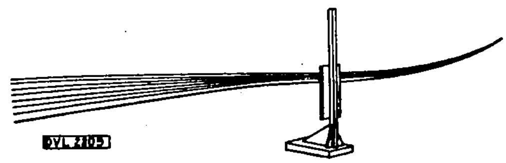


Figure 7.-Cable tensometer (taken from B & A of the WGL 1922, p.148, fig.25).

Figure 15.-Deflections of leading edge of wing (source ZFM, 1916, p.29, fig. 12).



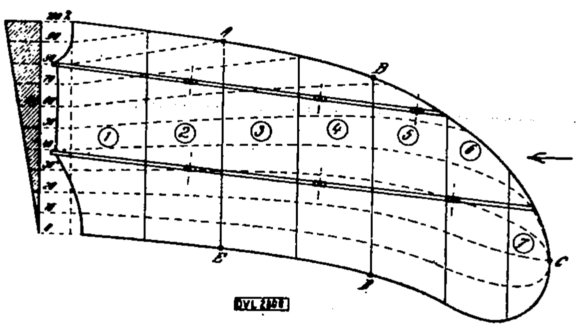


Figure 14.-Sketch of wing with load chart (source: ZFM, 1916, p.29, fig.9).

Figure 8.-Cable tensometer (Fig.26)

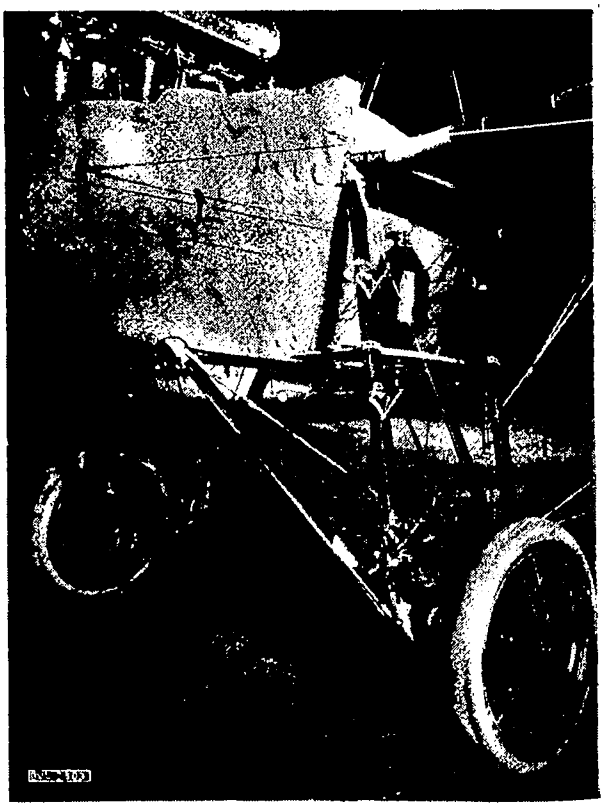


Figure 9.-Cable tensometer installed in Albatross Taube (Fig.27)

Figure 10.-Albatross BII (Fig.28)

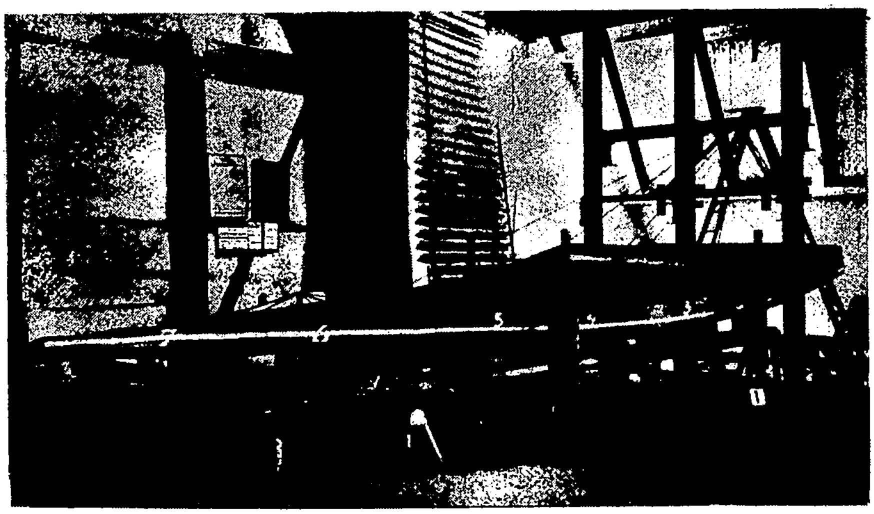


Figure 13.-Airplane wing under load test (from 1912/13 DVL yearbook, page 25).

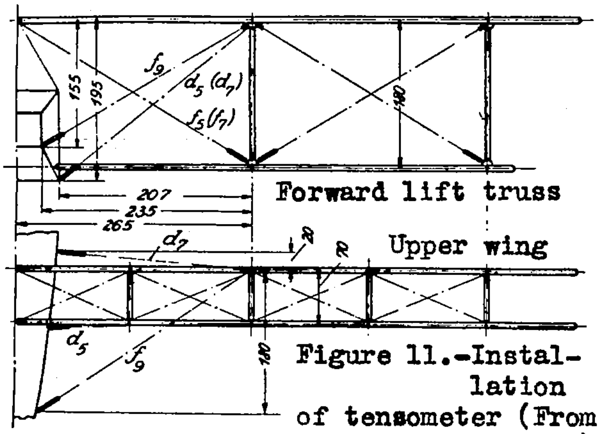


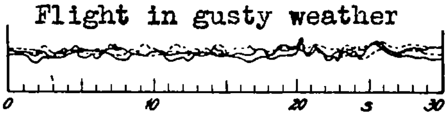
Figure 11.-Instal-
lation
of tensometer (From
TBI,p.84, table 28)

———	Front right	d5r
- - - - - -	Rear "	d7r
- - - - - -	Front left	d5l
———	Rear "	d7l

Lift wires d5l, d7l, d5r, d7r

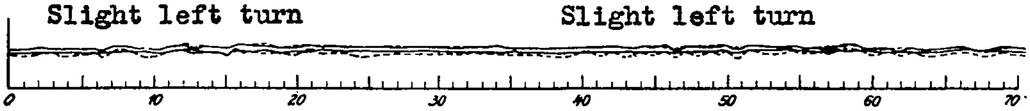
Flight in gusty weather

Flight No.4
June 14, 1914 a.m.
Pilot: Landmann



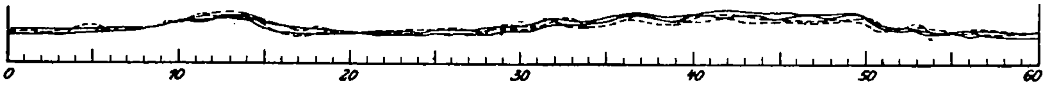
Level flight in calm weather

Slight left turn Slight left turn

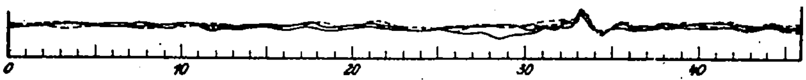


Flight No.5, June 15, 1914 a.m. Pilot: Landmann

Right turn Recovery Corkscrew left Recovery



Glide Pull out

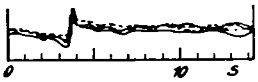


Flight No.12
June 17, 1914 a.m.

Flight No.12
June 17, 1914 a.m.
Pilot: Landmann

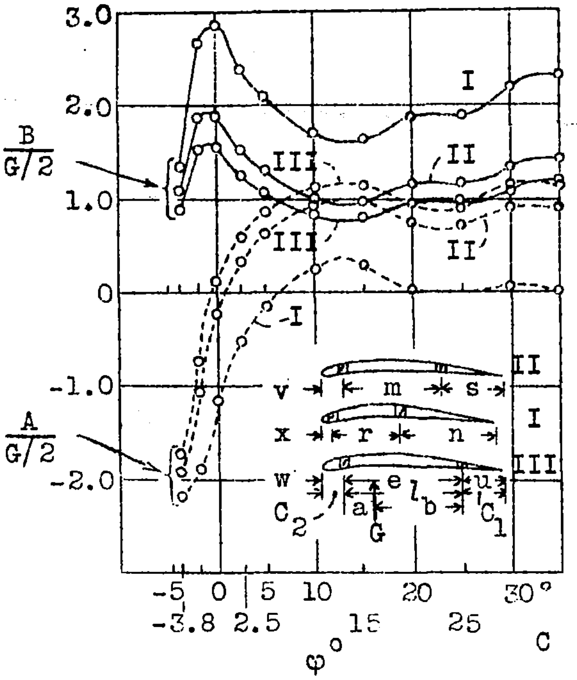
Albatros B II

Flight No.13
June 6, 1914 a.m.
Pilot: Landmann
Weight 942 kg
Glide & pull out



Flight No.14
June 17, 1914 a.m.
Pilot: v.Löss
Weight 1004 kg

Figure 12.-Indicator records for cable forces
(From TBI, p.84, table 25 & 26).



e,	1220
m,	1000
n,	990
r,	700
s,	650
u,	435
v,	230
w,	225
x,	110

Figure 16.-Distribution of total airplane weight A over the spars during glide at different path angles φ
A) load on front spar.
B) " " rear "

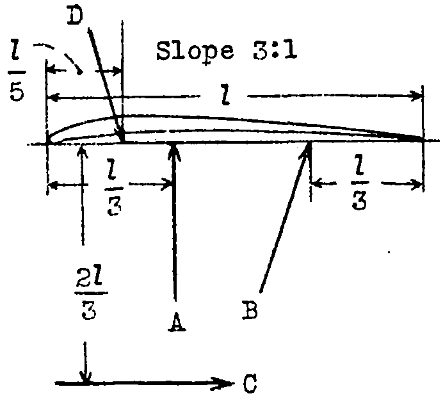


Figure 17.-Direction of air loads(source BLV, 1916, page 8).

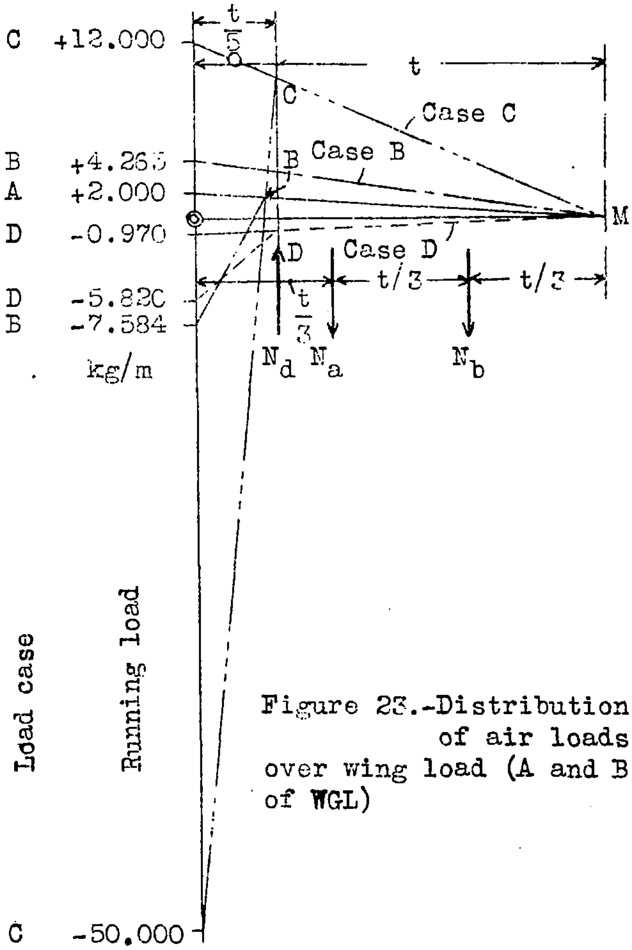


Figure 23.-Distribution of air loads over wing load (A and B of WGL)

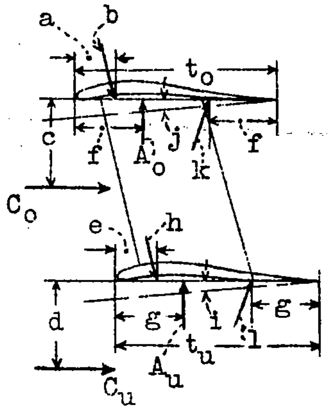


Figure 18.-Position of air loads.

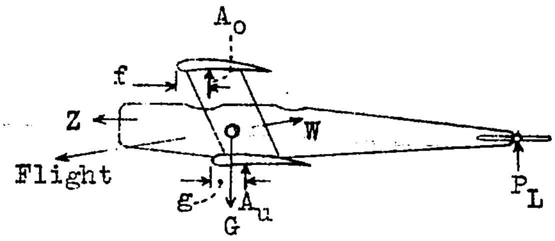


Figure 19.-A-case pull out.

a, $\frac{t_o}{5}$

b, D_o 4:1

c, 1.75 t_o

d, 1.75 t_u

e, $\frac{t_u}{5}$

f, $\frac{t_o}{3}$

g, $\frac{t_u}{3}$

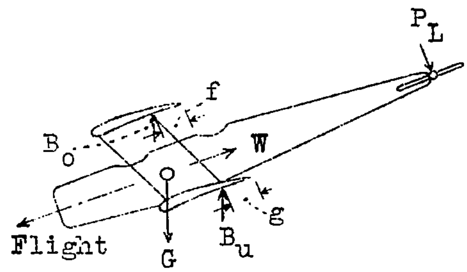


Figure 20.-B-case gliding flight.

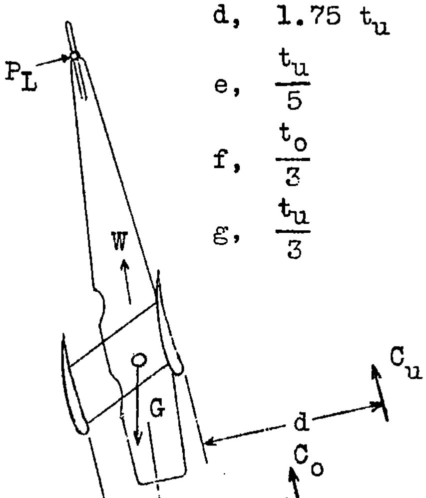


Figure 21.-C-case nose dive.

h, D_u 4:1

i, α_u

j, α_o

k, B_o 3:1

l, B_u 3:1

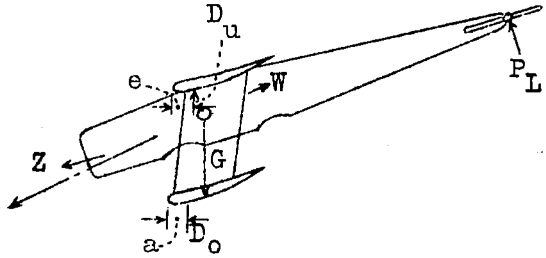


Figure 22.-D-case inverted flight.

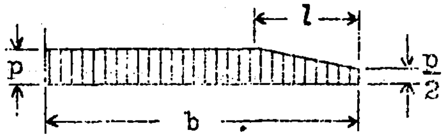


Figure 24.-Decrease of lift at wing tips (source. BLV, 1918, fig. 12).

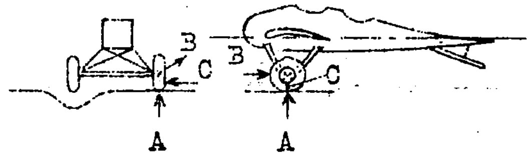


Figure 25.-Impact forces on landing gear (source BLV, 1918, fig. 15).

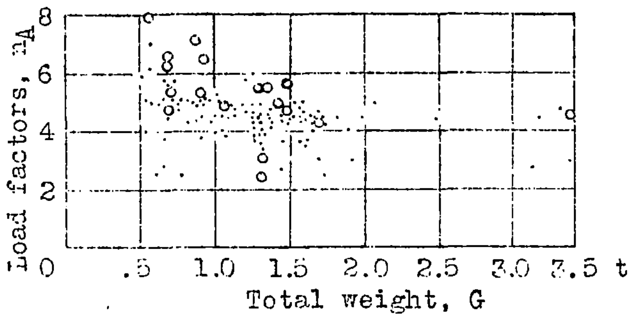


Figure 26.-Load case A

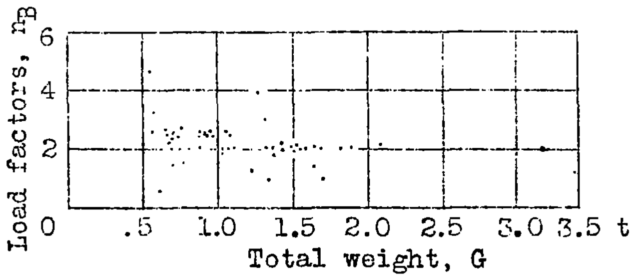


Figure 27.-Load case B

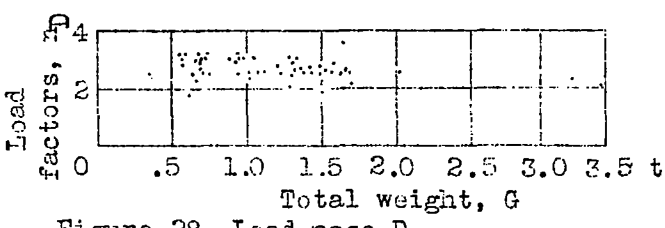


Figure 28.-Load case D

Figures 26,27,28.-Load tests on wings.

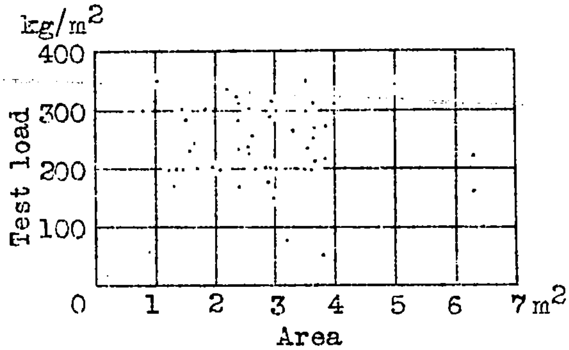


Figure 29.-Load test for stabilizer

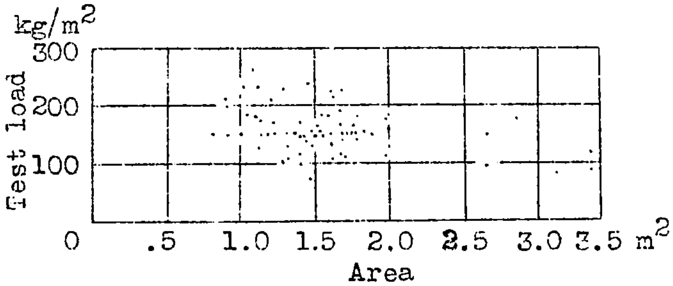


Figure 30.-Load test for elevator

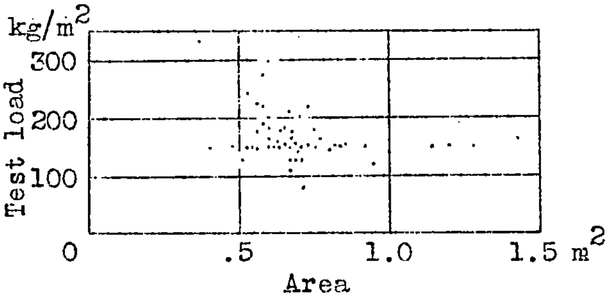


Figure 31.-Load test for rudder

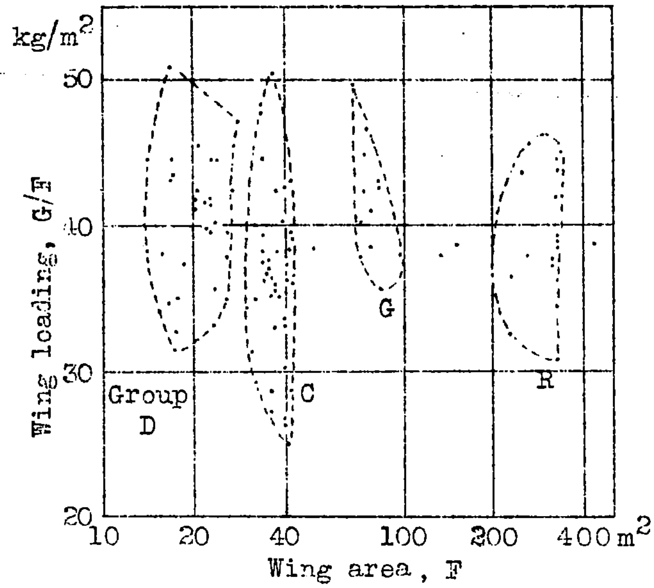


Figure 32.-Wing loading of German airplanes.

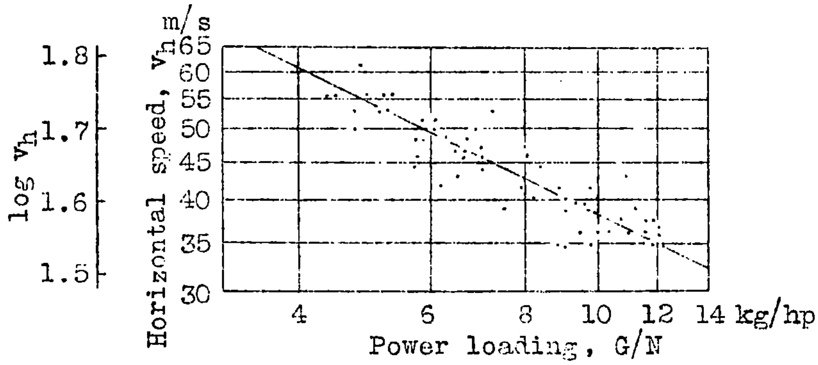


Figure 33.-Speed of German airplanes.

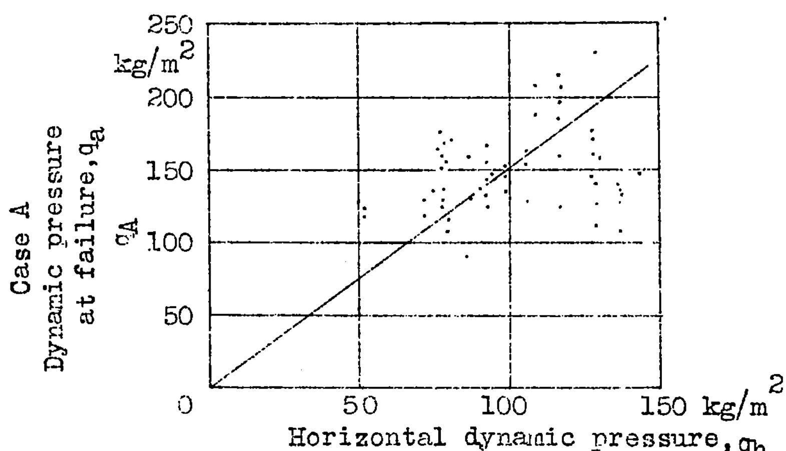


Figure 34.-Dynamic pressure, load case A